SHEET	1	4-086482-20			
TOTAL	63				
CODE		1500-9-1			
ISSUE DATE		6/26/03			
REVISION					
NO.	PAGE	DATE	NO.	PAGE	DATE
A	See note	3/25/03	H	See note	10/08/04
B	See note	4/03/03	J	See note	12/22/04
C	See note	6/26/03			
D	7,12,14-20,35	9/30/03			
E	See note	1/30/04			
F	See note	3/26/04			
G	See note	6/30/04			

ENGINEERING REPORT

FAA CONTRACT NO. DTFA03-02-C-00044
PHASE 3, CLIN 0003a (TASK 1) - DETAILED WORK PLAN

Distribution
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General Manager
Enabling Technologies

Revision J:
Revised to update Deliverable Status based on progress made during Quarter 9. Updated status of Task 9 Damage Characterization.

TABLE OF CONTENTS

APPENDIX A: Full-Scale Fatigue Test and Teardown Literature Review

The remaining Literature Reviews were moved to the Data Analysis and Inspection Capability reports at Q8.

LIST OF COMMON ACRONYMS

AD	Airworthiness Directive
BL	Butt Line (Aircraft Coordinate System)
CIC	Corrosion Inhibiting Compounds
ET	Eddy Current Transducer (NDT inspection)
DVI	Detailed Visual Inspection
EIFS	Equivalent Initial Flaw Size
EO	Engineering Order (Delta internal document for modification instructions)
FASTER	FAA's Full-Scale Aircraft Structural Test Evaluation and Research facility
FS	Fuselage Station (Aircraft Coordinate System)
GVI	General Visual Inspection
HFEC	High Frequency Eddy Current (NDT inspection)
JIC	Job Instruction Card (Delta internal document for routine inspection instructions)
LFEC	Low Frequency Eddy Current (NDT inspection)
MED	Multiple Element Damage
MFEC	Medium Frequency Eddy Current (NDT inspection)
MSD	Multiple Site Damage
SB	Service Bulletin
SDR	Service Discrepancy Report
SI	Special Inspection (Delta internal document for one-time and repetitive inspections)
SIF	Stress Intensity Factor
SRM	Structure Repair Manual
SSI	Structurally Significant Item
SSID	Supplementary Structural Inspection Document
WFD	Widespread Fatigue Damage
WL	Water Line (Aircraft Coordinate System)
WS	Wing Station (Aircraft Coordinate System)
UT	Ultrasonic Transducer (NDT inspection)

EXECUTIVE Summary

This report is a Detailed Work Plan of the Destructive Evaluation of N474DA. The purpose of this report is to provide a detailed overview of the project objectives and planning, including:

- an in-depth literature survey
- the overall technical approaches
- the responsibilities of participating organizations
- the scope and objectives of each major task area, with schedule, and expected results
- a Gantt chart showing all tasks and milestones.

This report is a living document, and will be updated each quarter in cooperation with the FAA to show updated timelines for each project task. In addition, the Detailed Work Plan will be updated to reflect any technical redirections received from the FAA.

CHAPTER 1. INTRODUCTION

This report is a Detailed Work Plan that supports Task 1 of FAA Contract DTFA03-02-C-00044. The purpose of this report is to provide a detailed overview of the project objectives and planning, including:

- an in-depth literature survey
- the overall technical approaches
- the responsibilities of participating organizations
- the scope and objectives of each major task area, with schedule, and expected results
- a Gantt chart showing all tasks and milestones.

All work required within Task 1 of the Statement of Work has been completed up to Quarter 9 (Phase 3). Revision H of this report satisfies the third of three deliverable requirements for CLIN 0003a.

CHAPTER 2. PROGRAM BACKGROUND AND OBJECTIVES

Background

In 1999, the AAWG published “Recommendations for Regulatory Action To Prevent Widespread Fatigue Damage in the Commercial Fleet” in response to FAA ARAC Tasking. The report contains the following pertinent conclusions and recommendations:

- With respect to research programs; report concluded that “additional research into the residual strength behavior of structure with MSD/MED should be conducted to supplement existing database”
- With respect to analytical assessment of MSD/MED; report concluded that “the analysis procedures used to characterize MSD/MED scenarios on airplanes needs careful correlation with test and service evidence”
- Recommended that “the FAA fund research ...,” and that “every effort should be made to make data from tests conducted in all research programs available at the earliest possible time...”
- Recommended that “funded research...involving lead crack link-up, should be accomplished as soon as possible to support the first round of audits due in three years”
- Recommendation for further research on application of equivalent initial flaw size methodologies
- Report identified 16 typical design detail susceptible to MSD/MED including fuselage longitudinal skin laps, frames and tear straps

Regulatory Guidance

Advisory Circular 91-56B, “Continuing Structural Integrity Program For Large Transport Category Airplanes,” recommends that “the time in terms of flight cycles/hours to the WFD average behavior in the fleet should be established.” This evaluation should include:

- Relevant and full-scale and component fatigue test data
- Teardown inspections
- The distribution of equivalent initial flaws, as determined from the analytical assessment of flaws found during fatigue test and/or teardown inspections regressed to zero cycles
- A distribution of fatigue damage determined from relevant fatigue testing and/or service experience

- A best estimate of undetected damage from inspection method used at end of test or during teardown inspections
- Progression of the crack distributions from the initial cracking scenario to the final cracking scenario should be developed...empirically from test or service fractographic data
- And, to determine the reliability of the Structural Modification Point...teardown inspections that could be done on structural components that have been removed from service
- Teardown evaluations are critical elements of effective WFD Prevention Programs

Service Findings in B727 Fuselage Longitudinal Lap Joints

In Dec 1998, multiple site damage fatigue cracks were found in the lap splice inner skin, lower fastener rows of B727 aircraft. The first finding was in the bilge along S-26L and was found while accomplishing adjacent repair (see Figure 1). Subsequent inspections of the 727 fleet showed that cracking was more common in the crown along S-4L and S-4R. This service event was not predicted by OEM full scale fatigue testing, since the joint was atypical in test article. Approximately 2000+ airplanes were manufactured with non-bonded lap joints (faying surface sealed).

Some observations from actual service failure investigation support additional research needs to fully address the following technical concerns:

- Every hole was cracked; many holes were cracked from both sides
- The longest cracks were in the center of the frame bay. Cracks at this location linked first to form a long lead crack in the center of the frame bay
- Subsequent link-ups with the MSD cracks hindered the lead crack's inherent tendency to flap. A sawtooth-shape shows the crack turning away from the fastener row, but then straightening once adjacent MSD was reached
- Holes were cracked in the same pattern in adjacent frame bays
- Frames at S-26L are shear-tied, with no tearstraps
- The crack had not flapped, and the arrest of the lead crack apparently was not permanent . Crack growth likely would resume if the crack were not detected

- Variability in crack initiation locations at hole edges affected detectability during subsequent inspection program.

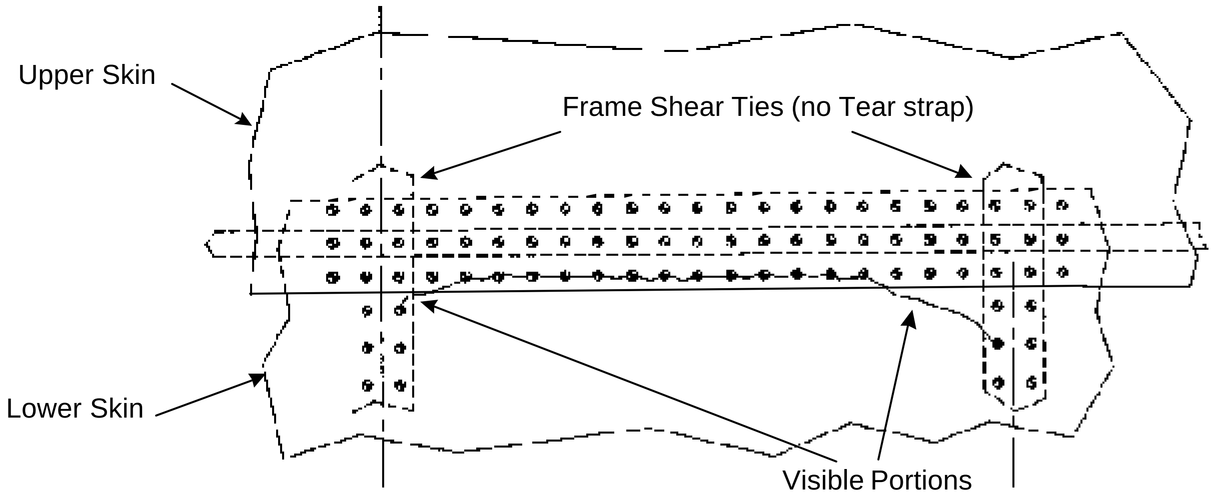


Figure 1: B727 Lap Joint Service Finding

Conclusions from the Literature Review

- Full-scale fatigue testing, teardown, inspection will provide valuable information on crack initiation, crack growth, distribution, linkup, fatigue life, and the effects of fatigue multiple-site damage/multiple element damage on the residual strength of fuselage structures. Aircraft destructive evaluation research provides the optimum way to validate and calibrate different crack initiation models.
- Full-scale fatigue test results can be used to establish a database on: as received full-scale fatigue testing sample inspection data, pre-teardown inspection data, after-tear-down inspection data, the initiation and growth of fatigue cracks from fractographic examination of the selected samples, fatigue MSD/MED initiation, distribution, linkup, and residual strength.
- Based on full-scale fatigue tests, nonlinear finite element analysis can be applied to predict the strain distributions and stress in the panel, crack initiation and crack growth, and to determine the effect of crack location and trajectory and material nonlinearity on the residual strength of the test panels. Note that linear FEA may yield erroneous results for typical airframe structure.

- NDI/E inspection can be validated and calibrated via aircraft teardown project. Different detailed NDI inspection programs can be established for aircraft teardown. Useful information about missed flaw sizes can be obtained from comparison with the teardown database.
- Equivalent Initial Flaw Size is the most promising approach to determine the crack initiation. An EIFS database can be established based on the full-scale fatigue testing. Eijkhout model, corrosion fatigue model, and Monte Carlo method are also useful for crack initiation study. The published initial fatigue crack initiation size for Al 2024-T3 ranges from $1x10^{-6}$ to $2.6x10^{-3}$ inch. The available (in the open literature) initial crack size for Al 7075-T6 ranges from $2.4x10^{-4}$ to $5x10^{-3}$ inch.
- The combination of actual service followed by component testing and destructive teardown offers the most comprehensive understanding of WFD precursors.

Technical Shortcomings from Literature Surveys

- Much of the published experimental and analytical research of MSD during the past decade has focused on lap joint upper skin cracking (e.g. Aloha 737 in 1988) in classical arrays (equal cracks). In full scale testing, these classical arrays are typically introduced through sawcuts or EDM notches, rather than through natural fatigue initiation.
- Accelerated full-scale fatigue tests often do not include environmental effects that may be important factors in crack initiation.
- Accelerated full-scale fatigue tests often do not account for true service spectrum. A truncated spectrum eliminates a high number of low-amplitude load cycles. These low-amplitude cycles are responsible for fretting and corrosion-fatigue interactions, which are important mechanisms in fatigue crack initiation.
- Published work often does not assess combinations of MSD with accidental damage (NDT detectable MSD combined with damage missed by visual maintenance programs). Little data exists on large damage capability concerning the combined effects of MSD/MED and accidental damage.
- Inner skin cracking raises new technical issues concerning detectability, panel bulging, and large crack arrest

– MSD may not be detected with a external visual inspection even after it has linked into a long lead crack. It can only be detected after the crack turns down from under the upper skin, typically at a 20” length.

– The upper skin restrains the lower skin even after the fatigue crack is long. Skin bulging along crack is reduced, and the crack tips remain loaded by hoop pressurization stresses.

– The tendency for pressurized skin to “flap” during failure is an important safety element in an aircraft fuselage. The dynamics of “flapping” may be significantly altered for a restrained lower skin. This is why the failure mode for MSD must be understood.

Understanding Widespread Fatigue Damage

This research program provides a unique opportunity to apply AAWG program assumptions and elements for WFD prevention, while adding to our knowledge base through investigation of the following technical issues:

- What is the extent of MSD cracking at the AAWG Structural Modification Point (SMP)? At operational link-up?
- How many growth cycles are there between SMP and operational link-up? Between operational link-up and failure?
- What extent of MSD exists before complex interactions among the cracks becomes significant?
- What influence does the type of loading (e.g. bending, shear) have on the resulting MSD array?
- What are the roles of hole quality and fastener fit in crack initiation?
- Does the presence of neighboring crack origins affect MSD crack growth?
- What factors can lead to crack initiation between holes?
- How effective are equivalent initial flaw size methodologies in predicting fatigue behavior of complex aircraft structure?

Areas for Supporting Research

During the course of this program, it is expected that avenues of further investigation into the understanding of Widespread Fatigue Damage will become evident. Those promising avenues that are beyond the scope of this 3-year project will be recommended to the academic community as areas for supporting research.

Parametric Rivet Residual Stress

The photomicrograph in Figure 2 shows the tilted rivet axis, degrees of cold work in the rivet material, axisymmetric bulging of the rivet shank, drilling defects, sealant and adhesive layers, and rivet contact areas in a typical lap joint lower row. Supporting research could concentrate on scientifically modeling dynamic wave effects of the riveting as well as 3-D FEA models capturing all the other contributing effects. It is recommended that the Georgia Institute of Technology be tasked with this research effort.

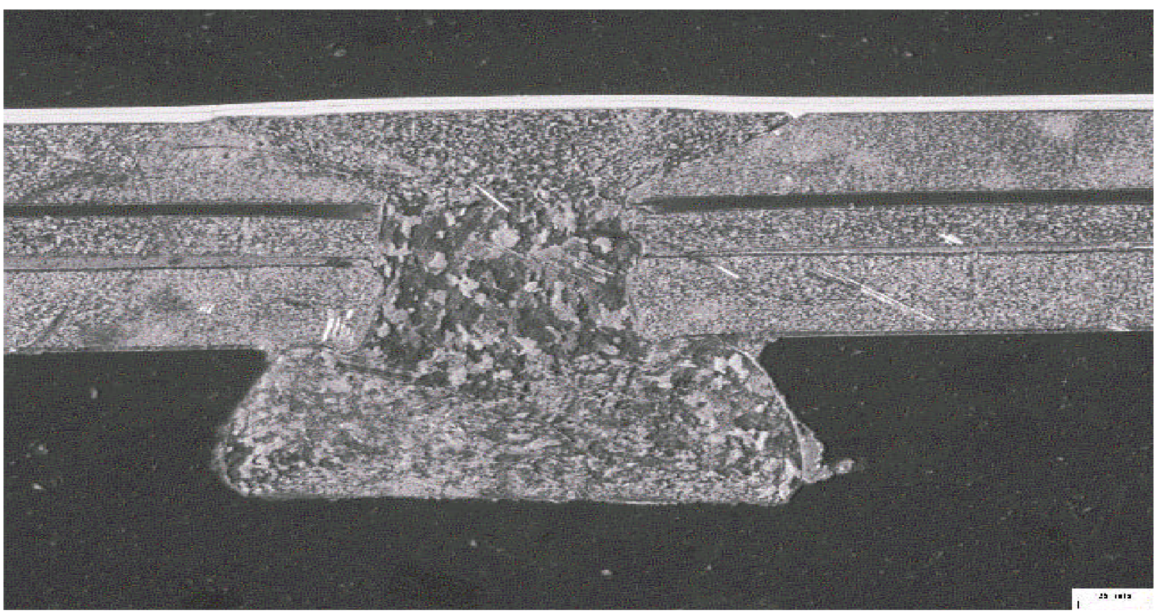


Figure 2: Typical Rivet Installation

Stress Intensity Solutions for Multiple Radiating Cracks

The photomicrograph in Figure 3 shows a typical finding from the Damage Characterization, which has yielded fuselage lap joint inner skin lower row rivet holes with multiple cracks radiating in many directions. Further research is needed to formulate stress intensity functions accurately simulating this typically observed damage state (multiple holes in a semi-infinite plate with multiple cracks radiating in different directions in a biaxial stress state). It is recommended that Mississippi State University be tasked with this research effort.

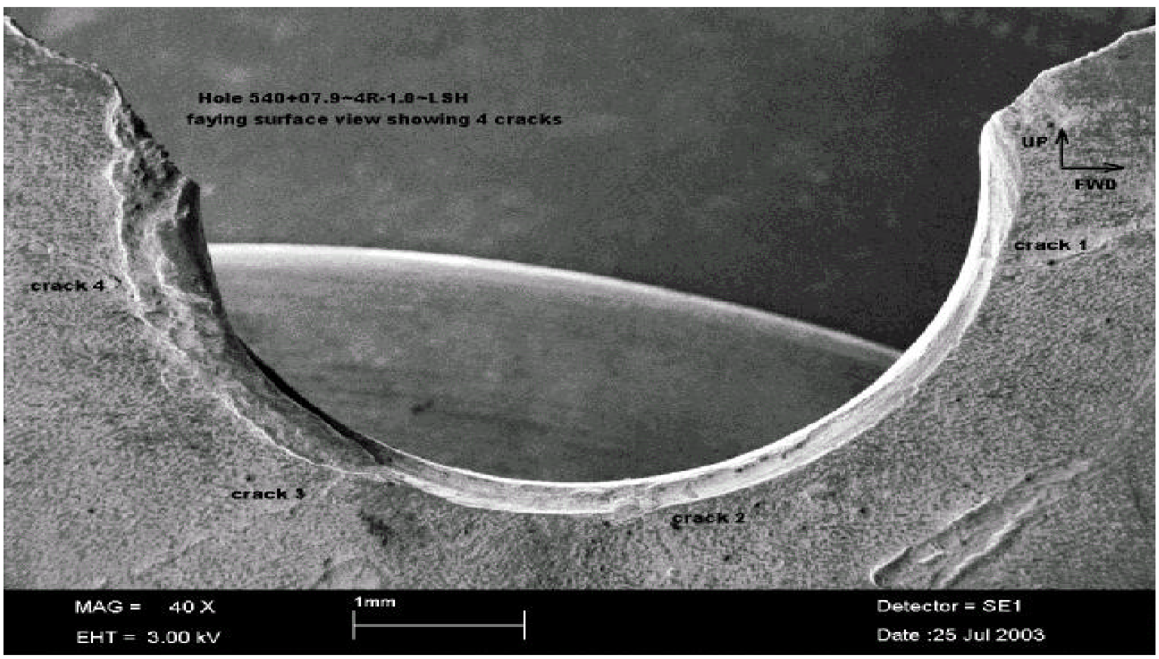


Figure 3: Multiple Cracks Radiating From Fastener Hole

CHAPTER 3. PROJECT SCHEDULE

Phase 1 Deliverables

(10/01/02 - 6/30/03)

Task	CLIN	Title	ECD
1	0001a(1)	Detailed Work Plan	Completed
	0001a(2)	Q2 Detailed Work Plan Update	Completed
	0001a(3)	Q3 Detailed Work Plan Update	Completed
2	0001b	Aircraft Information Report	Completed
3	0001c	Target Area Report	Completed
4	0001d	Field Inspection Report	Completed
5	0001e	Aircraft Test Specimens	Completed
	------	Specimens Removal Report	Completed
6	0001f	Pre-Teardown Inspection Report	Completed
7	0001g(1)	Preparation and Delivery of Test Panel FT2 to FAA Test Center	Completed
8	0001h(1)	FT2 Preliminary Test Plan	Completed
9	0001i	Q3 Damage Characterization Report	Completed
12	0001j	Initial Database	Completed

Phase 2 Deliverables

(7/01/03 - 3/31/04)

Task	CLIN	Title	ECD
1	0002a(1)	Q4 Detailed Work Plan Update	Completed
	0002a(2)	Q5 Detailed Work Plan Update	Completed
	0002a(3)	Q6 Detailed Work Plan Update	Completed
7	0001g(2)	Preparation and Delivery of Test Panel FT1 to FAA Test Center	Completed
	0001g(3)	Preparation and Delivery of Test Panels FT3 and FT4 to FAA Test Center	Completed
8	0002	Test Plan Analysis	Completed
	0002c(1)	FT2 Final Test Plan	Completed
	0002c(2)	FT1 Final Test Plan	Completed
	0002c(3)	FT3 Final Test Plan	Completed
	0002c(4)	FT4 Final Test Plan	Completed
9	0002d(1)	Q4 Damage Characterization Report	Completed
	0002d(2)	Q5 Damage Characterization Report	Completed
	0002d(3)	Q6 Damage Characterization Report	Completed
10	0002e(1)	Q4 Data Analysis Report	Completed
	0002e(2)	Q5 Data Analysis Report	Completed
	0002e(3)	Q6 Data Analysis Report	Completed

Phase 3 Deliverables

(4/01/04 - 12/31/04)

Task	CLIN	Title	ECD
1	0003a(1)	Q7 Detailed Work Plan Update	Completed
	0003a(2)	Q8 Detailed Work Plan Update	Completed
	0003a(3)	Q9 Detailed Work Plan Update	Completed
9	0003b(1)	Q7 Damage Characterization Report	Completed
	0003b(2)	Q8 Damage Characterization Report	Completed
	0003b(3)	Q9 Damage Characterization Report	Completed
10	0003c(1)	Q7 Data Analysis Report	Completed
	0003c(2)	Q8 Data Analysis Report	Completed
	0003c(3)	Q9 Data Analysis Report	Completed
11	0003d(1)	Q7 Inspection Capability Report	Completed
	0003d(2)	Q8 Inspection Capability Report	Completed
	0003d(3)	Q9 Inspection Capability Report	Completed

Phase 4 Deliverables

(1/01/05 - 9/30/05)

Task	CLIN	Title	ECD
1	0004a(1)	Q10 Detailed Work Plan Update	3/25/05
	0004a(2)	Q11 Detailed Work Plan Update	6/24/05
	0004a(3)	Q12 Detailed Work Plan Update	9/23/05
9	0004b	Q10 Damage Characterization Report	3/25/05
10	0004c(1)	Q10 Data Analysis Report	3/25/05
	0004c(2)	Q11 Data Analysis Report	6/24/05
	0004c(3)	Q12 Data Analysis Report	9/23/05
11	0004d(1)	Q10 Inspection Capability Report	3/25/05
	0004d(2)	Q11 Inspection Capability Report	6/24/05
	0004d(3)	Q12 Inspection Capability Report	9/23/05
12	0004e	Technical Report - Draft	6/24/05
	0004f	Technical Report – Final (Phases 1-4)	9/23/05
	0004g	Final Database (Phases 1-4)	7/22/05

Phase 5 Deliverables

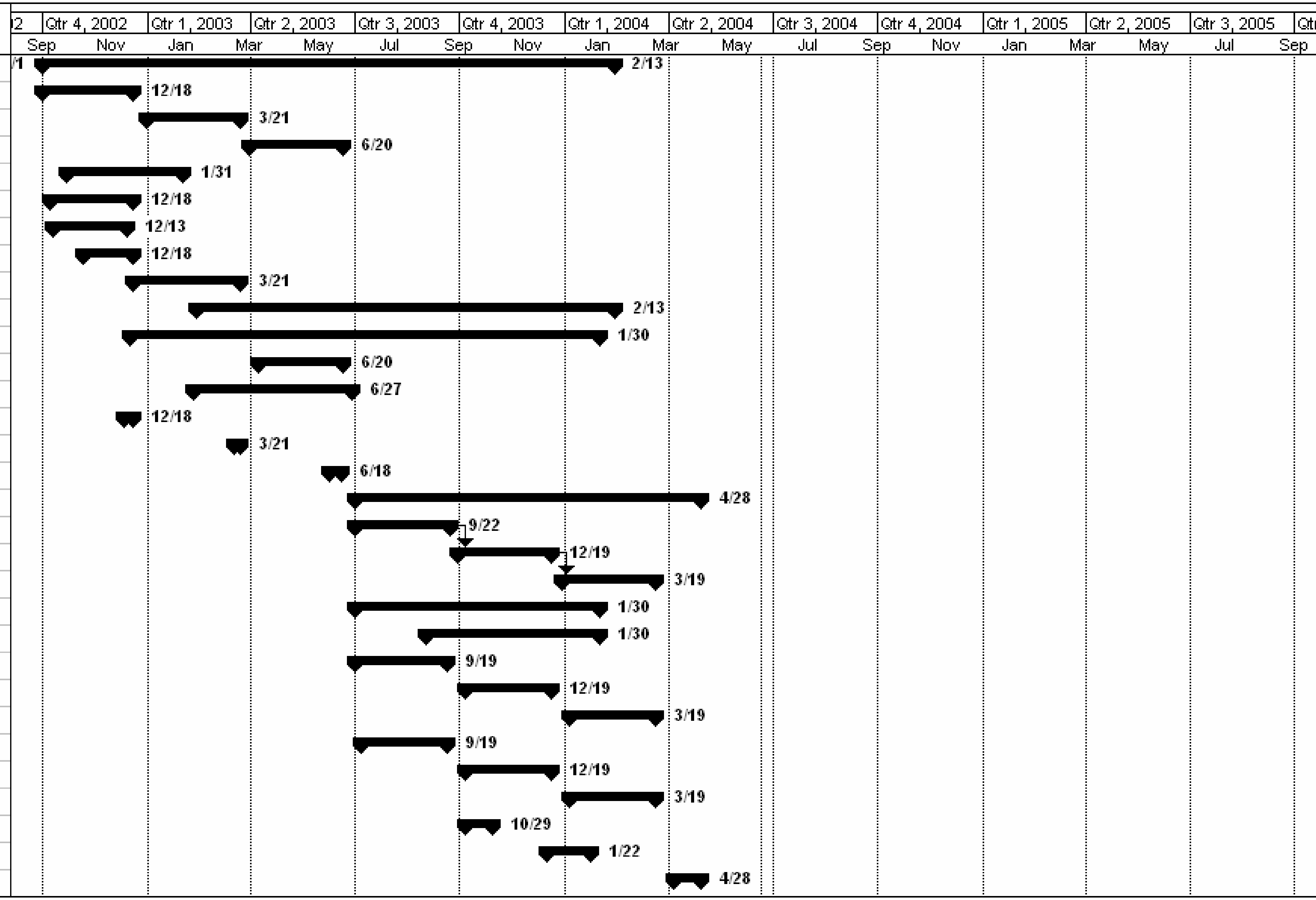
(9/30/05 – 3/30/06)

Task	CLIN	Title	ECD
1	0005a(1)	Q13 Detailed Work Plan Update	12/31/05
	0005a(2)	Q14 Detailed Work Plan Update	3/31/06
9	0005b	Q13 Damage Characterization Report	12/31/05
10	0005c	Q13 Data Analysis Report	12/31/05
12	0005d	Technical Report Addendum - Draft	2/28/06
	0005e	Technical Report Addendum - Final	3/31/06
	0005f	Final Database	3/31/06

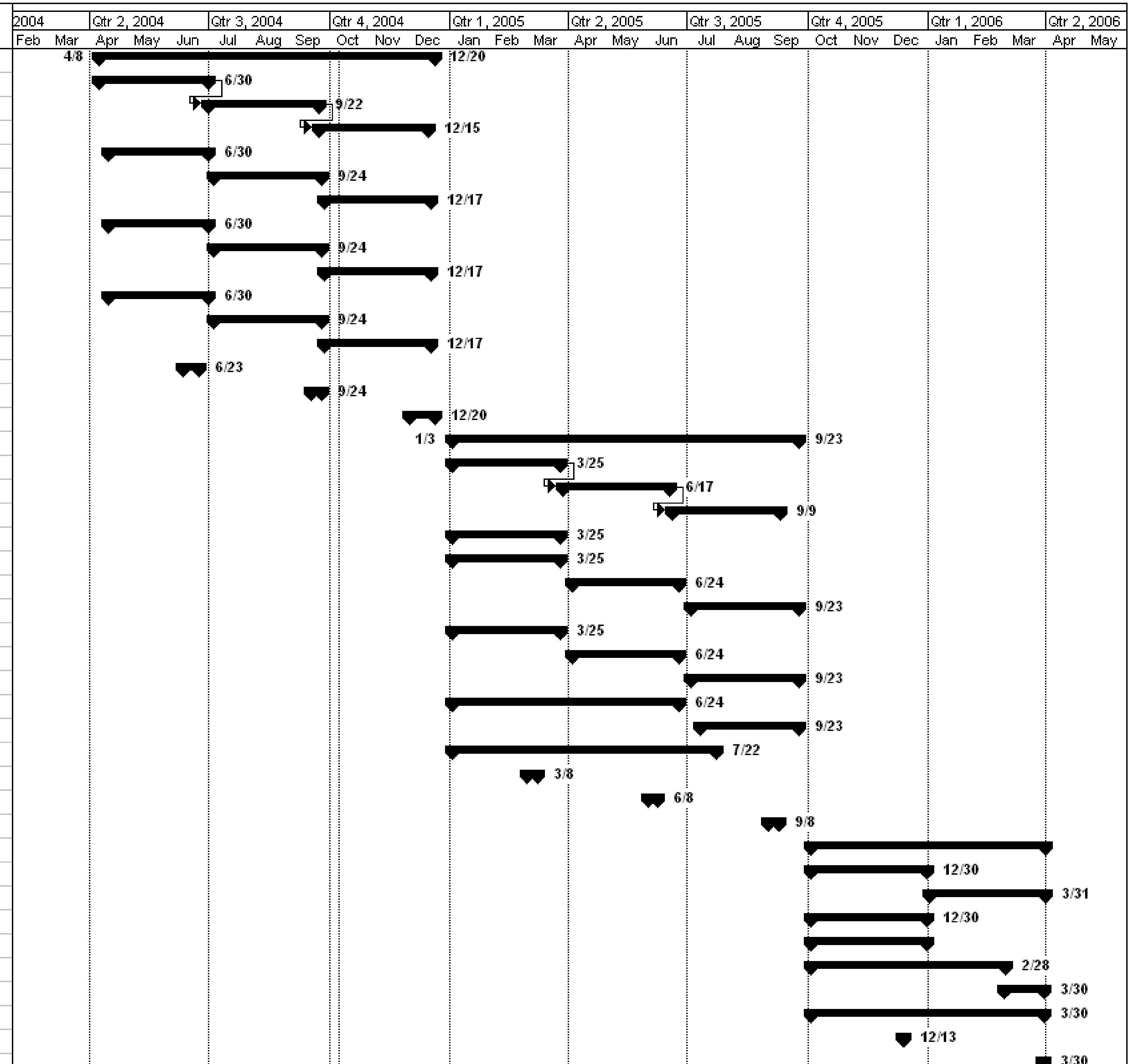
Note: The contract revision required to extend the performance period and add these deliverables is pending.

Task Area Gantt Chart

ID	Task Name
7	Phase 1
8	0001a(1): Initial DWP
11	0001a(2): DWP Q2 Update
13	0001a(3): DWP Q3 Update
15	0001b: Aircraft Information Report
20	0001c: Target Area Report
27	0001d: Field Inspection Report
33	0001e: Specimen Removal Report
39	0001f: Pre-teardown Inspection Report
44	0001g: Delivery of Fuselage Panels Report
59	0001h: Preliminary Test Plan
77	0001i: Phase 1 Damage Characterization Report
82	0001j: Initial Database
85	Q1 meeting
89	Q2 meeting
93	Q3 meeting
97	Phase 2
98	0002a(1) DWP Q4 Update
100	0002a(2): DWP Q5 Update
102	0002a(3): DWP Q6 Update
104	0002b: Test Plan Analysis
111	0002c: Final Test Plan
116	0002d(1): Q4 Damage Characterization Report
118	0002d(2): Q5 Damage Characterization Report
120	0002d(3): Q6 Damage Characterization Report
122	0002e(1): Q4 Data Analysis Report
124	0002e(2): Q5 Data Analysis Report
126	0002e(3): Q6 Data Analysis Report
128	Q4 meeting
131	Q5 meeting
134	Q6 meeting

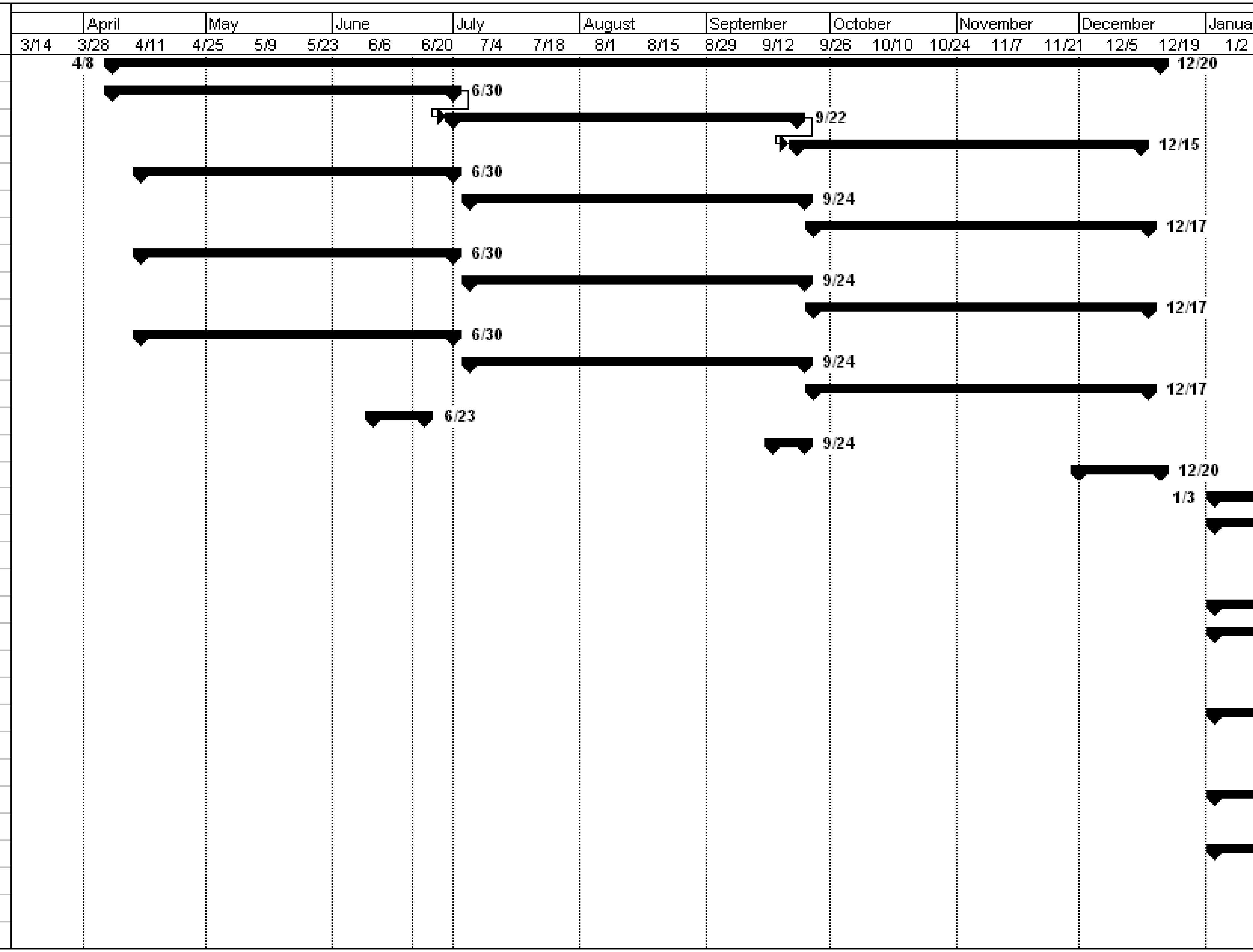


ID	Task Name	
137	Phase 3	4/8 – 12/20
138	0003a(1) DWP Q7 Update	6/30
140	0003a(2): DWP Q8 Update	9/22
142	0003a(3): DWP Q9 Update	12/15
144	0003b(1): Q7 Damage Characterization Report	6/30
146	0003b(2): Q8 Damage Characterization Report	9/24
148	0003b(3): Q9 Damage Characterization Report	12/17
150	0003c(1): Q7 Data Analysis Report	6/30
152	0003c(2): Q8 Data Analysis Report	9/24
154	0003c(3): Q9 Data Analysis Report	12/17
156	0003b(1): Q7 Inspection Capability Report	6/30
158	0003d(2): Q8 Inspection Capability Report	9/24
160	0003d(3): Q9 Inspection Capability Report	12/17
162	Q7 meeting	6/23
165	Q8 meeting	9/24
168	Q9 meeting	12/20
171	Phase 4	1/3 – 9/23
172	0004a(1) DWP Update Q10	3/25
174	0004a(2): DWP Q11 Update	6/17
176	0004a(3): DWP Q12 Update	9/9
178	0004b: Q10 Damage Characterization Report	3/25
180	0004c(1) Q10 Data Analysis Report	3/25
182	0004c(2) Q11 Data Analysis Report	6/24
184	0004c(3) Q12 Data Analysis Report	9/23
186	0004d(1) Q10 Inspection Capability Report	3/25
188	0004d(2) Q11 Inspection Capability Report	6/24
190	0004d(3) Q12 Inspection Capability Report	9/23
192	0004e: Technical Report - Draft	6/24
194	0004f: Technical Report - Final	9/23
196	0004g: Final Database	7/22
198	Q10 meeting	3/8
202	Q11 meeting	6/8
206	Q12 meeting	9/8
210	Phase 5	
211	0005a(1) DWP Update Q13	12/30
213	0005a(2) DWP Update Q14	3/31
215	0005b Q13 Damage Characterization Report	12/30
217	0005c Q13 Data Analysis Report	
219	0005d Technical Report Addendum - Draft	2/28
221	0005e Technical Report Addendum - Final	3/30
223	0005f Final Database	3/30
225	Q13 meeting	12/13
227	Q14 meeting	3/30



Phase 3 Gantt Chart

ID	Task Name
137	Phase 3
138	0003a(1) DWP Q7 Update
140	0003a(2): DWP Q8 Update
142	0003a(3): DWP Q9 Update
144	0003b(1): Q7 Damage Characterization Report
146	0003b(2): Q8 Damage Characterization Report
148	0003b(3): Q9 Damage Characterization Report
150	0003c(1): Q7 Data Analysis Report
152	0003c(2): Q8 Data Analysis Report
154	0003c(3): Q9 Data Analysis Report
156	0003b(1): Q7 Inspection Capability Report
158	0003d(2): Q8 Inspection Capability Report
160	0003d(3): Q9 Inspection Capability Report
162	Q7 meeting
165	Q8 meeting
168	Q9 meeting
171	Phase 4
172	0004a(1) DWP Update Q10
174	0004a(2): DWP Q11 Update
176	0004a(3): DWP Q12 Update
178	0004b: Q10 Damage Characterization Report
180	0004c(1) Q10 Data Analysis Report
182	0004c(2) Q11 Data Analysis Report
184	0004c(3) Q12 Data Analysis Report
186	0004d(1) Q10 Inspection Capability Report
188	0004d(2) Q11 Inspection Capability Report
190	0004d(3) Q12 Inspection Capability Report
192	0004e: Technical Report - Draft
194	0004f: Technical Report - Final
196	0004g: Final Database
198	Q10 meeting
202	Q11 meeting
206	Q12 meeting



CHAPTER 4. TASK AREA DETAILS

Task 1: Detailed Work Plan / Program Review

Contract Statement of Work

The contractor, in coordination with the FAA, will develop a detailed work plan (DWP) outlining the overall technical approaches, responsibilities of participating organizations, scope and objectives of each major task area, schedule, and expected results. The DWP shall include an in-depth literature survey to document procedures, guidelines, lessons learned, and results from previous airplane destructive evaluations and fatigue testing. The DWP outlines the overall technical approaches, responsibilities of participating organizations, scope and objectives of each major task area, schedule, and expected results. A Gantt chart showing all tasks and milestones will be included and shall be updated by the contractor through out the period of performance of the contract. This DWP shall incorporate any technical re-directions received as approved by the FAA. The DWP shall be modified and submitted to the FAA on a quarterly bases.

Contract Deliverables

Phase 1	0001a: Detailed Work Plan & quarterly updates
Phase 2	0002a: Quarterly updates to the Detailed Work Plan
Phase 3	0003a: Quarterly updates to the Detailed Work Plan
Phase 4	0004a: Quarterly updates to the Detailed Work Plan
Phase 5	0005a: Quarterly updates to the Detailed Work Plan

Task 2: Selection Of Candidate Aircraft

Contract Statement of Work

The contractor, in coordination with the FAA, will select one Federal Aviation Regulation (FAR) Part 25 certified aircraft near its DSG. The aircraft must hold up to 120 passengers and be representative of revenue-service passenger aircraft currently in the domestic fleet of FAR Part 121 aircraft with at least 700 currently in service. The aircraft must have a well-documented and accessible service history. A prerequisite for the aircraft chosen is that the entire usage in terms of flight types, mix and hours be known. The aircraft shall have at least 75% of the 16 WFD susceptible structure defined in the Appendix A.

Contract Deliverable

Phase 1 0001b: Aircraft Information Report, with history of aircraft including aircraft usage maintenance records, service difficulty reports, FAA Airworthiness Directives (AD), etc.

Scope and Objectives

I. Verify that candidate aircraft N474DA (727-200) satisfies the SOW prerequisites.

A. Determine 727-200 certification basis

B. Determine current status of the worldwide and FAR 121 727 fleets.

C. Verify the presence of WFD susceptible structure in the 727-200 design

II. Document maintenance history and present in an accessible format.

A. Document the history of routine maintenance, including block checks, letter checks, and HMV's (D-checks). Compile history into an accessible format.

B. Document the history of compliance with Airworthiness Directives and Boeing Service Bulletins. AD's and S/B's that affect the specimens to be removed will merit a more detailed summary. Compile history into an accessible format.

C. Document the history of Delta-designed repairs by compiling Engineering Repair Authorizations. Repairs that affect the specimens to be removed will merit a more detailed summary. Compile engineering records into an accessible format.

D. Document the history of Delta-designed alterations through Supplemental Type Certificates. STC's that affect the specimens to be removed will merit a more detailed summary. Compile engineering records into an accessible format.

E. Query the FAA database to gather history of Service Discrepancy Reports. Highlight those reports which affect the specimens to be removed.

III. Determine entire usage of N474DA in terms of flight types, mix and hours from delivery in 1974 to retirement in 1998. These parameters will change throughout the life of the aircraft.

A. Determine flight cycles and flight hours versus date.

B. Determine usage data in terms of cycles/day and hours/cycle

C. Determine typical flight profile, including CG location, typical altitude, and typical load factor.

Responsibilities

Delta is responsible for all work within this task

Results

I. N474DA is acceptable as a candidate aircraft. See Aircraft Information Report

Task 3: Selection Of Aircraft Specimens

Contract Statement of Work

In the base program, the contractor, in coordination with the FAA, will select four or more of the following fuselage structures susceptible to multiple site damage (MSD) and/or multiple element damage (MED) for examination:

a) Longitudinal Skin Joints, Frames, and Tear Straps of Pressure Vessel (MSD/MED)

b) Circumferential Joints and Stringers of Pressure Vessel (MSD/MED)

c) Lap joints with Milled, Chem-milled or Bonded Radius (MSD)

d) Fuselage Frames (MED)

e) Stringer to Frame Attachments of Fuselage (MED)

f) Shear Clip End Fasteners on Shear Tied Fuselage Frames (MSD/MED)

For the fuselage structures sections, the total area of all selected details shall not be less than 400 sq feet:

a) with not less than 90 linear feet of skin joints susceptible to MSD and

b) with not less than 10 recurrent structural elements susceptible to MED and

c) with not less than 250 individual details (e.g. rivet sites) susceptible to MSD or MED.

In addition, the contractor, in coordination with the FAA, will select six or more structures for use later in the optional program (if exercised by the FAA) that are susceptible to multiple site damage (MSD) and/or multiple element damage (MED) for examination:

a) Over Wing Fuselage Attachments (MED)

b) Latches and Hinges of Non-plug Doors (MSD/MED)

c) Aft Pressure Dome Outer Ring and Dome Web Splices (MSD/MED)

d) Skin Splice at Aft Pressure Bulkhead (MSD)

e) Abrupt Changes in Web or Skin Thickness — Pressurized or Unpressurized Structure (MSD/MED)

f) Window Surround Structure (MSD, MED)

g) Skin at Runout of Large Doubler (MSD)—Fuselage, Wing or Empennage

h) Wing or Empennage Chordwise Splices (MSD/MED)

i) Rib to Skin Attachments (MSD/MED)

j) Typical Wing and Empennage Construction (MSD/MED)

Each of these features is illustrated in Attachment J-1 Appendix A [of the SIR]. The target areas for destructive analysis in the base program shall include four fuselage panels suitable for extended fatigue cycling and residual strength tests using the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility located at the FAA William J. Hughes Technical Center, Atlantic City International Airport, NJ. A description of the facility and the requirements for the panels is attached as Attachment J-2 Appendix B [of the SIR].

<u>Contract Deliverable</u>

Phase 1 0001c: Target Area Report, listing selected areas including engineering drawings, location on aircraft, and selection justification.

<u>Scope and Objectives</u>

I. Select fuselage skin areas for the Base Program

A. Select and justify 4 fuselage skin areas to modify into FASTER panels

1. Determine panel sizes such that FASTER frame spacing, stringer spacing, and overall size requirements are met.

2. Calculate circumferential and longitudinal stress under flight loads and pressurization along the fuselage.

3. Locate panels on aircraft to place S-4 near the panel centerline, and a circumferential joint within a tested section. Allow ample margin for finish trimming after removal.

B. Locate remaining skin panels such that the specified minimums for linear feet, square feet, and individual details are met.

II. Select fuselage, wing, and empennage structures for use in optional program

A. Determine which structures have a high susceptibility to MSD/MED to justify selection

1. Find repeated details as shown in SIR Appendix A

2. Calculate spanwise stress under flight loads along the wing lower surface

3. Cross-reference Principal Structural Elements as identified by Boeing 727 SSID

B. Verify that the specified minimums MSD/MED susceptible details are met.

III. Find or generate engineering drawings of selected structure

A. Create new drawing of 4 panels for FASTER modification.

B. Document Boeing drawings for all other structures

<u>Responsibilities</u>

The selection of structures in this Task is to be done in cooperation with the FAA. In particular, FAA technical input is required to:

- Make final determination as to which areas of the removed crown panels are most suitable for FASTER testing, based on the state of initial damage determined in Task 4

<u>Results</u>

All work required by this task has been completed. The Target Area Report will be submitted to the FAA 12/18/02.

Task 4: Conduct Field Inspections

Contract Statement of Work

Prior to removing sections defined in Task 3, a general visual field inspection, including photographs of the sections, should be made by the contractor in coordination with the FAA to catalog and fully document the condition of the sections. The contractor shall label the sections with boundaries and identification marks to indicate the location and orientation of the section with respect to the aircraft. Both visual and NDI for cracks, corrosion and disbonds of the selected structure shall be conducted by the contractor using procedures based on the OEM's recommended standard practice and directed inspection requirements (service bulletins or airworthiness directives). NDI procedures shall be modified to allow data acquisition of signal response data to be analyzed later.

Contract Deliverable

Phase 1 0001d: Field Inspection Report, illustrating location and condition of structure prior to removal with photographs and drawings.

Scope and Objectives

I. Prepare for visual and NDI for cracks, corrosion and disbonds

A. Determine which Service Bulletins and AD's are applicable to the Base Program structure

B. Compile instructions (job cards, process standards, etc.) for inspection procedures based on S/B's, AD's, and Boeing standard practice.

C. Modify standard NDI procedures to allow data acquisition of signal response data to be analyzed later

II. Conduct visual and NDI for cracks, corrosion and disbonds

A. Remove interior, wiring, and systems in Sections 41 through 46 for as necessary for access

B. Conduct field NDI and DVI inspections the Base Program structure

III. Prior to removing specimens, document condition

A. Conduct general visual field inspection of aircraft, with special attention to Base and Optional structures

B. Label the sections with boundaries and identification marks to indicate location and orientation

C. Photograph specimens to show condition and to show relative location on airframe.

Note that the tasks to be performed on the additional structures are only those required for proper documentation in anticipation of later work. A general visual field inspection, including photographs of the sections, should be made to catalog and fully document the condition of the sections, and the sections will be labeled with boundaries and identification marks to indicate the location and orientation of the section with respect to the aircraft. However, NDI and S/B and AD directed inspections will not be performed to the additional structures.

Responsibilities

Delta inspectors and engineers are responsible for all work within this task.

Results

All work required by this task has been completed. The Field Inspection Report will be submitted to the FAA 12/18/02.

The field inspections showed that MSD cracking is present within the S-4L and S-4R lap joints.

Task 5: Removal Of Specimens

Contract Statement of Work

The contractor shall remove the selected sections from the aircraft without damaging those structural details identified for analysis. Care must be taken by the contractor to provide adequate support for the structure during removal to prevent overloading and damage of the sections. The skins and substructure of the four fuselage panels designated for testing in the FASTER facility must not be breached or otherwise made unacceptable for testing in the FASTER facility. The contractor shall be responsible for the safe shipment of sections to the testing and analysis sites. The contractor shall be responsible for damage associated with removal and shipping of the specimens.

Contract Deliverable

Phase 1 0001e: Aircraft Test Specimens to testing and analysis sites

Scope and Objectives

I. Prepare for aircraft disassembly

A. Determine shoring plan and disassembly sequence to remove the selected sections without damaging them due to overload, drop, puncture, etc.

B. Write instructions for shoring and disassembly

II. Remove selected sections from aircraft

A. Mark cut-lines on aircraft.

B. Provide engineering guidance during the installation and removal of shoring

C. Provide engineering guidance for section removal, including choices for crane equipment, strapping methods, and ground storage.

D. Document and photograph disassembly.

III. Package specimens for shipment to ensure safe shipment of sections to the testing and analysis sites

A. Design and/or coordinate shipping containers

B. Coordinate and provide engineering guidance for truck loading and shipment.

Responsibilities

Delta engineers are responsible for all work within this task.

Expected Results

All work required by this task has been completed. Base and Optional Program Specimens have been safely removed and delivered to the Delta TOC. The Specimen Removal Report will be submitted to the FAA 12/18/02.

Base Program panels will continue for further work as described in the following tasks. Optional Program structures are FAA property after removal, but they will be stored by Delta for the duration of the contract.

Task 6: Conduct Pre-Teardown Inspections

<u>Contract Statement of Work</u>

After removal and delivery of the sections to the testing and analysis site, a general visual inspection with photographs of the sections should be made by the contractor in coordination with the FAA to document the as-received condition. The contractor shall catalog and document detailed descriptions of the sections removed including engineering drawings showing geometric dimensions, photographs taken before and after removal, exact location and orientation with respect to the aircraft, and procedure for removal.

The contractor using the procedures in Task 4 shall conduct both visual inspection and NDI for cracks, corrosion and disbonds in the structure. In addition, prior to removing fasteners, the contractor shall conduct and document NDI and visual measurements of crack length, rivet head and tail diameters, countersink fits, and other joint assembly parameters.

<u>Contract Deliverable</u>

Phase 1 0001f: Pre-Teardown Inspection Report, illustrating location and condition of panels in the as-received condition with photographs and drawings.

<u>Scope and Objectives</u>

I. Prepare the panels for inspection. Unpack and stabilize panels

II. Conduct visual inspection to document the as-received condition

A. Take photographs

B. Create engineering drawings showing geometric dimensions of the panels

C. Catalog and document detailed descriptions in the database (see Task 12)

D. Conduct and document NDI

E. Conduct visual measurements with either a magnifying glass or 20x microscope. Document measurements of crack length, rivet head and tail diameters, countersink fits, and other joint assembly parameters into the database.

III. Create engineering drawings of the panels, showing structural dimensions, NDT indications, and any damage found.

IV. Compile findings into the Pre-teardown Inspection Report and the database

Responsibilities

Delta inspectors and engineers are responsible for all work related to this task.

Expected Results

It is expected that most cracks will be in the lower fastener row of the longitudinal lap joints. But, for this task, inspections must be done in the upper and lower lap joint rows, both sides of the circumferential butt joints, along the frames, and at the stringer clips.

All work required by this task has been completed. See the Pre-Teardown Inspection Report for more information.

Task 7: Preparation Of Panels

Contract Statement of Work

Four fuselage panels removed from the selected aircraft shall be tested using the FAA FASTER facility. The objectives of the extended testing are to: 1) propagate and extrapolate the state of damage beyond the one DSG, 2) characterize and document the state of damage through real-time NDI, high magnification visual measurements, and post-test destructive evaluation of fracture surfaces; and 3) correlate analysis methods to determine crack initiation and detection, first link-up and residual strength. In order to separate cracks from extended fatigue testing and from service conditions, an underload marker band spectrum should be applied prior to the fatigue testing. Three subtasks are included:

Preparation of Panels: The contractor shall prepare four fuselage test panels for proper fit and testing in the FASTER fixture as defined in Appendix B. This includes attaching reinforced load attachment doublers for the longitudinal, hoop, and frame end load assemblies for both fatigue and residual strength tests. The FAA shall provide geometric specifications and requirements of all loading attachments. The prepared specimens shall be shipped by the contractor to the FASTER facility located at the FAA William J. Hughes Technical Center.

Contract Deliverable

Phase 1 0001g: Preparation, Testing & Delivery of Fuselage Panels at the FASTER facility located at the FAA William J. Hughes Technical Center

Scope and Objectives

- The first work of this task will be a trip by Delta engineers to the FASTER test facility for detailed familiarization with the fixture.
    - To understand mechanical fittings and interfaces between the FASTER load frame and the panels, required for this task
    - To understand the capabilities and limitations of the FASTER load frame related to load application (See Task 8)
- Preparation of the four test panels includes:
    - Developing the final design for the test panels, in close cooperation with the FAA

    - Issuance of engineering instructions for modifications from removed specimens to test panels, including final engineering drawings of the test panels.
    - Precision cutting of skin edges and both ends of 6 frames and 7 stringers
    - Fabricating and attaching shear fittings to the 4 edges
    - Fabricating and fitting load attachments to both ends of 6 frames and 7 stringers
- Shipping the test panels includes:
    - Modifying the shipping containers from Task 5 to fit the test panels
    - Loading the panels into the containers (e.g., banding and bubble-wrap)
    - Carefully loading the truck

<u>Responsibilities</u>

Delta engineers and mechanics are responsible for the objectives listed above for the modification of the panels. In addition, the test objectives listed in the Statement of Work will be incorporated by Delta into the Test Plans covered in Task 8.

The FAA is responsible for

- providing an on-site briefing on the mechanical fittings and interfaces between the FASTER load frame and the panels
- providing geometric specifications and requirements of all loading attachments

While all work required to prepare the panels for testing has been completed, engineering work in support of the FASTER tests is ongoing. See the Test Plan Analysis and the 4 Final Test Plans for more information.

Task 8: Development Of Test Plan

Contract Statement of Work

The contractor, in coordination with the FAA, shall develop a Test Plan. The plan shall include loading spectra (test operating pressure, hoop and longitudinal loads, frame loads and cycle frequency), visual inspection and NDI requirements, test schedule, pre-test predictions with anticipated number of cycles, strain gage layouts and specifications, data collection requirements, and other information that are required to assure successful test results. The test plan shall outline the overall technical approaches, responsibilities of participating organizations, scope and objectives of each major task area, schedule, and expected results. The test plan shall be submitted to the FAA one month prior to testing for approval. The test plan shall include a Gantt chart showing all tasks and milestones and engineering drawings of test panels. The Gantt chart shall be updated monthly through out the period of performance of the test program..

Contract Deliverables

Phase 1	0001h:	Preliminary Test Plan
Phase 2	0002b:	Test Plan Analysis, including spectrum development
	0002c:	Final Test Plan

Based on discussion at the Q2 Quarterly Review Meeting, the schedule for the Task 8 deliverables has been staggered to be consistent with the test panel schedule in Task 7. Separate Test Plan Reports will be submitted for each panel. The FT2 Preliminary Test Plan was submitted coincident with delivery of the FT2 test panel. Submittal of the Final Test Plans for FT1, FT3, and FT4 will coincide with delivery of the applicable test panel.

Scope and Objectives

Development of the Preliminary Test Plan includes:

- Visit FASTER test facility for detailed familiarization to understand the capabilities and limitations of the FASTER load frame (completed 3/18/03).
- Determine objectives and designated panel for each of four tests, and coordinate with the FAA (completed 3/27/03, see Preliminary Test Plan Overview).

- Outline the overall technical approach, and responsibilities of participating organizations
- Determine stress exceedance plot representative of 727 service
  - Exceedance data will account for flight types, flight mix and flight hours flown by Ship 474
  - Exceedance data will be based on Delta Shuttle mission
- Accomplish order-of-magnitude crack growth prediction for planning purposes.
- Determine load spectra to apply to test fixture actuators
  - Determine crack growth rate in service based on flight, pressurization, and ground loads.
  - The FASTER load spectrum is a constant amplitude or composite spectrum that produces equivalent crack growth to service.
  - The test spectrum for each panel may be different.
- Determine strain gauge locations and technical requirements for data acquisition system.
- Determine schedule for visual inspection and standard and emerging NDI during the test.

Completion of the Test Plan Analysis includes

- Detailed crack growth simulation to predict fatigue test behavior.
  - The simulation is a panel level crack growth analysis that includes every lower-row fastener hole
- The simulation can be done with a common stress intensity function that is valid for every hole.
- geometry of skin hole with loaded fastener
- Adjacent FEA crack interaction effects
- Models of FASTER panels used to determine
  - load distributions among skin, stringer, and frames
  - mid-bay skin bulging due to stiffening at frames and stringers
  - FASTER boundary condition effects

The Final Test Plan shall include all information required to assure successful test results:

- visual inspection and NDI requirements, strain gage layouts and specifications, data collection requirements, engineering drawings of test panels;
- loading spectra (test operating pressure, hoop and longitudinal loads, frame loads and cycle frequency);

- the overall technical approaches, responsibilities of participating organizations, scope and objectives of each major task area;
- test schedule, and a Gantt chart showing all tasks and milestones, to be updated monthly through out the test program;
- pre-test predictions with anticipated number of cycles and expected results.
- The Final Test Plan shall be submitted to the FAA one month prior to testing for approval.
- Final Test Plans for all four panels are completed and delivered at Q5. Testing to the first panel (FT2) is scheduled to begin Apr 2004.

<u>Test Overview</u>

- Tests are planned to panels which have no NDI indications at the S-4 lap joint. This allocation allows all holes with NDI indications to be characterized without further crack propagation, ensuring accurate crack information for subsequent Probability of Detection studies.
- All planned tests have the same profile. Tests will begin using a representative fatigue spectrum. Near the end of the test, when visual MSD is present, the cyclic load will be increased to the residual strength spectrum and the panel will be cycled until failure.
    - The "Fatigue Spectrum" is a simplified constant amplitude or composite spectrum derived to produce crack growth rates and MSD distributions representative of those seen in service.
    - The "Residual Strength Spectrum" is a constant amplitude spectrum based on the critical residual strength requirement of 14 CFR 25.571(b)/JAR 25.571 (b). Applying this spectrum will result in crack growth rates that are faster than those seen in service. However, the MSD state at panel failure is by definition a critical MSD state for the panel.
- Additional details are available in the Test Plan Analysis.

Strain Surveys

The Final Test Plans for each of the four panels includes a strain survey at the fatigue condition prior to the start of the extended fatigue testing. The objectives of the strain surveys are:

- Create a high-fidelity strain map of the test panel at Design Service Goal, before additional damage has initiated. This map will be the baseline for comparison with later strain measurements as MSD develops.
- Validate the FEA models used to develop the test plans, and ensure that the panel is behaving as expected.
- Optimize the FEA models as necessary to improve correlation with the strain measurements. The FEA models' boundary conditions are particularly important to this correlation.

As FT2 was the first panel to be tested, the strain gage surveys in advance of fatigue testing were particularly exhaustive. Initial surveys were conducted at 5 psi internal pressure to ensure that all internal loads were relatively low. These tests have been completed, with the conclusion that there is good correlation between the predicted and measured strains in all significant areas for lap joint MSD. Therefore, the "go-ahead" to conduct strain surveys at the 8.9 psi fatigue load has been relayed to the FAA. See the Q6 Data Analysis Report for more details.

Guidance for Monitor vs Repair

As of Q7, FT2 has accumulated more than 17,200 cycles with no definitive indication of lap joint MSD. However, cracking has been detected in two other areas: near the bend radius of several stringer clips, and at the first skin to stringer attachment inboard of the outermost tearstraps, near the panel reinforcement doublers. Both of these are considered consequences of FASTER D-Box testing, so these cracks will be repaired, stop-drilled, or disregarded, whichever is more convenient (see the Q7 Data Analysis for more details). These actions are in contrast to the test plan for MSD in the lap or circumferential joints, where crack growth will be closely monitored and documented.

In general, the cracks of greatest interest are those located within the "area of interest" box that is separated from the edge reinforcing doublers by one frame or stringer bay. Cracks in the skin, frames, or stringers within the area of interest will likely be monitored and documented. Cracks in other areas will be addressed on a case-by-case basis, and are likely to be addressed as test consequences. This is particularly true for cracks resulting from the FASTER test environment which are not typical of service. An assessment of the effectiveness of D-Box testing to match the service environment will be included in the “lessons learned” of this test program.

For reference - From 14 CFR Part 25, §25.571 Damage tolerance and fatigue evaluation of structure:

(b) The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical maneuvering conditions specified in § 25.337 at all speeds up to Vc and in § 25.345.

(2) The limit gust conditions specified in § 25.341 at the specified speeds up to VC and in § 25.345.

(3) The limit rolling conditions specified in § 25.349 and the limit unsymmetrical conditions specified in §§ 25.367 and 25.427 (a) through (c), at speeds up to VC.

(4) The limit yaw maneuvering conditions specified in § 25.351(a) at the specified speeds up to VC.

(5) For pressurized cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in paragraphs (b)(1) through (4) of this section, if they have a significant effect.

(ii) The maximum value of normal operating differential pressure (including the expected external aerodynamic pressures during 1 g level flight) multiplied by a factor of 1.15, omitting other loads.

For reference - From JAR 25.571:

(b) Damage-tolerance (fail-safe) evaluation.

The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and (if available) service experience. Damage at multiple sites due to prior fatigue exposure must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical manoeuvring conditions specified in JAR 25.337 up to VC and in JAR 25.345.

(2) The limit gust conditions specified in JAR 25.341 at the specified speeds up to VC and in JAR 25.345.

(3) The limit rolling conditions specified in JAR 25.349 and the limit unsymmetrical conditions specified in JAR 25.367 and JAR 25.427(a) through (c), at speeds up to VC.

(4) The limit yaw manoeuvring conditions specified in JAR 25.351 at the specified speeds up to VC.

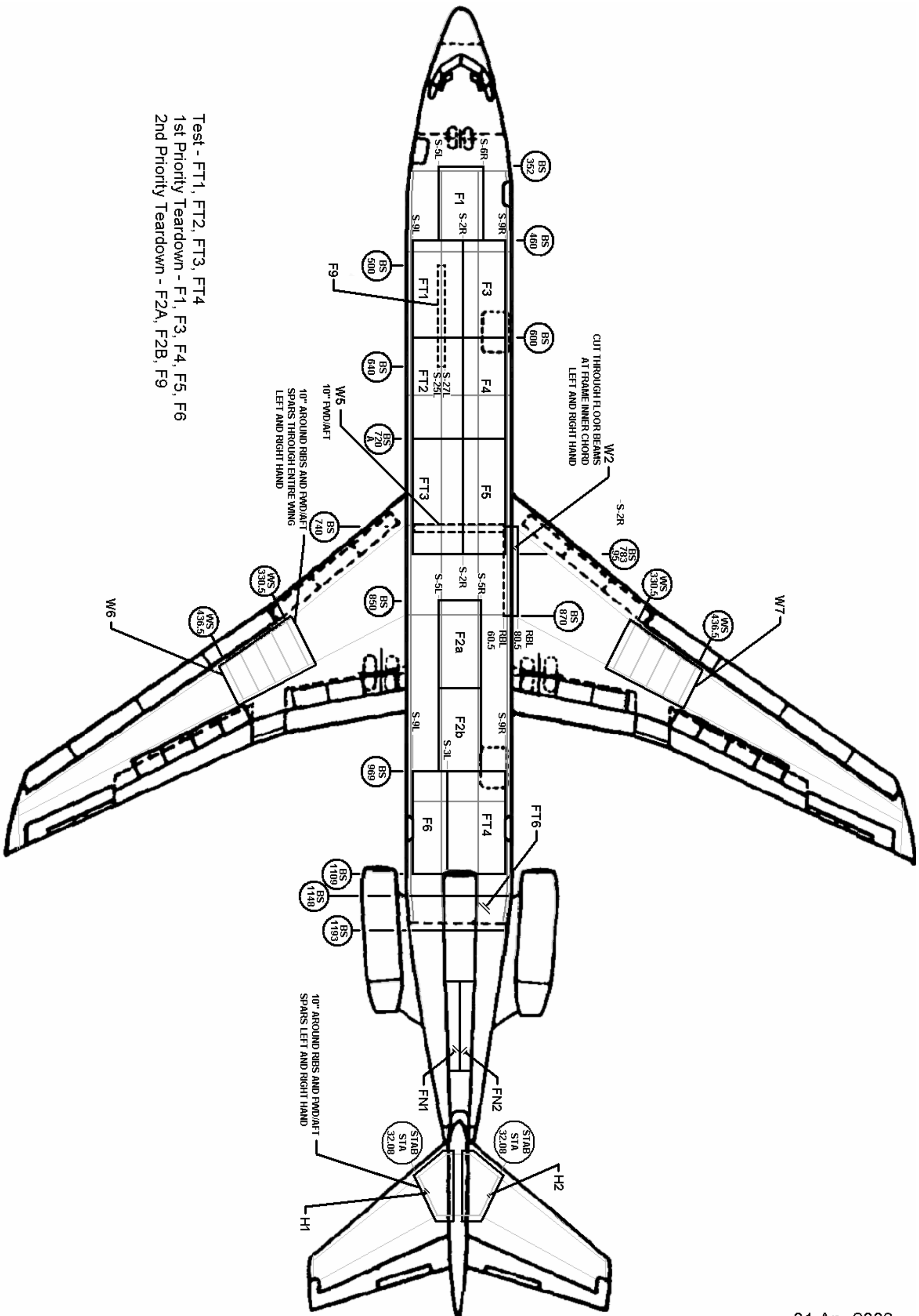
(5) For pressurised cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in subparagraphs (b)(1) to (b)(4) of this paragraph if they have a significant effect.

(ii) The maximum value of normal operating differential pressure (including the expected external aerodynamic pressures during 1 g level flight) multiplied by a factor of 1·15 omitting other loads.

Test Matrix

FASTER Test Matrix			
Test Sequence	Panel Number	Initial MSD Condition	Test Summary
2	Panel FT1 FS 460- FS 600 S-2R to S-9L (Left side)	None detected	• Load cycle until Visual MSD using representative Fatigue Spectrum. • Load cycle from Visual MSD to failure using Residual Strength Spectrum (i.e., critical condition from 14 CFR 25.571(b))
1	Panel FT2 FS 600- FS 720A; S-2R to S-9L (Left side)	None detected	• Load cycle until Visual MSD using representative Fatigue Spectrum. • Load cycle from Visual MSD to failure using Residual Strength Spectrum (i.e., critical condition from 14 CFR 25.571(b))
4	Panel FT3 FS 720A-FS 783 S-2R to S-9L (Left side)	None detected	• Load cycle until Visual MSD using representative Fatigue Spectrum. • Load cycle from Visual MSD to failure using Residual Strength Spectrum (i.e., critical condition from 14 CFR 25.571(b))
3	Panel FT4 FS 969- FS 1109 S-3L to S-8R (Right Side)	None detected	• Load cycle until Visual MSD using representative Fatigue Spectrum. • Load cycle from Visual MSD to failure using Residual Strength Spectrum (i.e., critical condition from 14 CFR 25.571(b))


Test - FT1, FT2, FT3, FT4
1st Priority Teardown - F1, F3, F4, F5, F6
2nd Priority Teardown - F2A, F2B, F9
F1
F3
F4
F5
F2a
F2b
F6
FT1
FT2
FT3
FT4
F9
W2
CUT THROUGH FLOOR BEAMS AT FRAME INNER CHORD LEFT AND RIGHT HAND
W5
10" FWD/AFT
10" AROUND RIBS AND FWD/AFT SPARS THROUGH ENTIRE WING LEFT AND RIGHT HAND
W6
W7
FT6
FN1
FN2
H1
H2
10" AROUND RIBS AND FWD/AFT SPARS LEFT AND RIGHT HAND
BS 352
BS 460
BS 500
BS 600
BS 640
BS 720 A
BS 740
BS 783 95
BS 850
BS 870
BS 969
BS 1109
BS 1148
BS 1193
WS 330.5
WS 436.5
STAB STA 32.08
RBL 60.5
RBL 80.5
S-5L
S-5R
S-6R
S-2R
S-9L
S-9R
S-27L
S-25L
S-3L

01 Ap. 2003

Task 9: Damage Characterization

Contract Statement of Work

The contractor shall characterize the state-of-damage of all sections removed including the panels to be tested in Task 7. This includes quantifying the extent of fatigue cracking, corrosion, faying surface fretting fatigue, and structural disbonding.

a. Fasteners Holes with Detected Cracks: Fastener holes with visually detect cracks and those whose NDI resulted in a signal response exceeding a specified threshold will be destructively examined. This threshold will be specified to ensure that no less than 150 fastener sites are destructively examined for the base program.

1. Fastener Removal and Destructive Examination: The contractor, in coordination with the FAA, shall remove selected fasteners from the structure and perform microscopic examination and bolt-hole eddy current inspections. The contractor shall split open select fastener hole to reveal the crack surfaces and then conduct a fractographic characterization. The crack shape and size should be cataloged and documented, however, evidence of crack initiation could be destroyed due to the fastener removal process. The fastener hole surfaces should be examined to determine and document the extent of damage due to the fastener removal process.

2. Destructive Examination with Fastener Intact: The contractor, in coordination with the FAA, shall select fastener holes to split open to reveal the crack surfaces with the fastener intact. The recommended method is a three-point bend straining technique developed at NASA Langley Research Center (LaRC). The technique, illustrated in Appendix C, exposes crack surfaces without damaging the fastener hole surface. The contractor shall then perform fractographic examinations to identify, catalog and document crack initiation sites and mechanisms, crack shapes and sizes, and quality of the fastener hole surface (compare with that from a removed fastener in Task 8a1 above).

b. Fastener Holes with No Crack Detection: For an arbitrary number of details (not less than 50 for the base program) with no visual or NDI evidence of cracking, a destructive examination shall be conducted. The objective is to determine if there are small cracks that were not detected by visual inspection and NDI. A subset of fastener holes will be identified for stripping and etching to highlight and enhance cracks for high magnification (at least 20X) visual inspection. The remaining fastener holes shall be destructively examined per task 8a1 and 8a2 above.

c. Other Damage Modes: the contractor shall examine Structure with conditions known to precipitate cracking. Such conditions include structural disbonding, corrosion and faying surface fatigue. The contractor, in coordination with the FAA, shall select structure based on NDI and visual inspection results to measure and quantify the extent of corrosion, faying surface fatigue, and structural disbonds (tear-straps, lap joints, etc).

d. Reconstruct Crack History: Using an scanning electron microscope (SEM) or equipment with equivalent resolution and standard fractographic methods including conduct striation counts, the contractor shall empirically reconstruct crack growth histories on no less than 50 characteristic cracks for the base program. Crack fronts and profiles should be defined by at least three positions having beach marks.

Contract Deliverables

Phase 1: 9a: Damage Characterization Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 9.

Phase 2: 9b: Damage Characterization Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 9.

Phase 3: 9c: Damage Characterization Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 9.

Phase 4: 9d: Damage Characterization Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 9.

Scope and Objectives

The primary objectives of this task are well explained within the above statement of work. The scope of the damage characterization includes :

- Within at least one frame bay, fully characterize the damage state of all structures to catalogue the types of damage present (Visual and SEM characterization).
    - Determine extent of damage in upper and lower skin, stringers, frames, and tearstraps.
    - Document the micro- and macro-scale conditions contributing to crack initiation.
- On several test or teardown panels, measure the crack length, crack orientation, surface quality and fastener parameters along the lower skin lower fastener row holes (Visual characterization and SEM striation counts).
    - Document conditions contributing to crack initiation to validate engineering models relating fastener fit, fracture morphology, and crack initiation/EIFS.
    - Document evidence of crack interaction.
- At all inspection locations, measure crack length parameters of detected cracks (Visual characterization).
    - Individual crack lengths are needed for NDI Probability of Detection studies
    - Distribution of crack lengths is needed to determine crack arrays for crack growth and residual strength models.
- Satisfy all SOW requirements, including the characterization of 50 holes with no crack indications.
    - Examination of holes in the upper skin lower row, stringer, and tearstraps will determine the level of damage in those details which are not NDI inspected by current programs.
    - Examination of all lower skin lower row holes across a frame bay will provide small crack data for the crack array.

In addition to the primary objectives, several lines of investigation have been developed based on findings from the damage characterizations to date. These lines include:

- Investigate the effect of the poor rivet installation as a cause for more than two cracks per hole.
- Investigate the cracks that are oriented in the fuselage hoop direction.
- Investigate the retardation effects, if any, of the multiple cracks in the same hole.
- Determine if fractographic indications can be found that resulted from N474DL's mission change from main line to shuttle aircraft.
- Determine if load shedding from the lower row fastener hole cracking has caused cracking in the lower skin in the stringer row of holes.

Phase 1 Results

Prototype disassembly and visual characterization of large MSD cracks in S-4R lower row between fuselage stations 540 and 560. See Q3 Damage Characterization Report.

Phase 2 Results

Characterization via visual microscopy of MSD along S-4R, FS 480 – FS 600 and FS 720C – 720D. Striation counts for selected cracks within this range. See Q6 Damage Characterization Report.

Phase 3 Expected Scope

- Full characterization (visual microscopy and SEM) of all significant lower row cracks, S-4R, FS 540 to 560.
- Characterization of FT2 cracking outside the test area in support of FASTER testing.
- Initial characterization of lap joint MSD from FT2 FASTER testing. Evaluation of the efficacy of the specified marker band sequence, in consideration of future testing.

Phase 4 Expected Scope

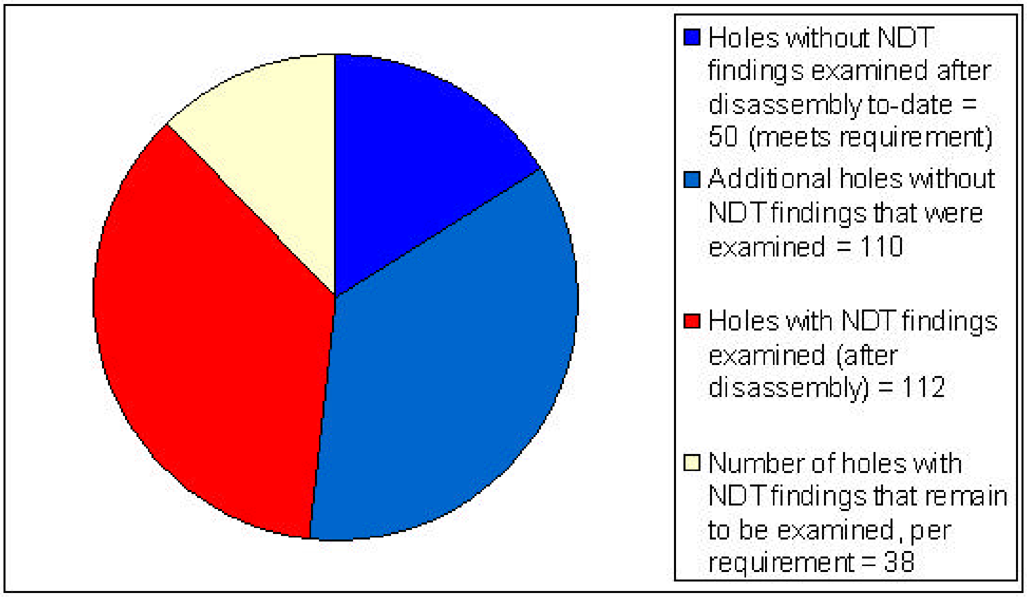
- Characterization via visual microscopy of remaining MSD from Pre-Teardown Inspection.
- Characterization of lap joint MSD from FT1 FASTER testing.

Phase 5 Expected Scope

- Characterization of lap joint MSD from FT3 and FT4 FASTER testing.

Responsibilities

Delta engineers are responsible for all work within this Task



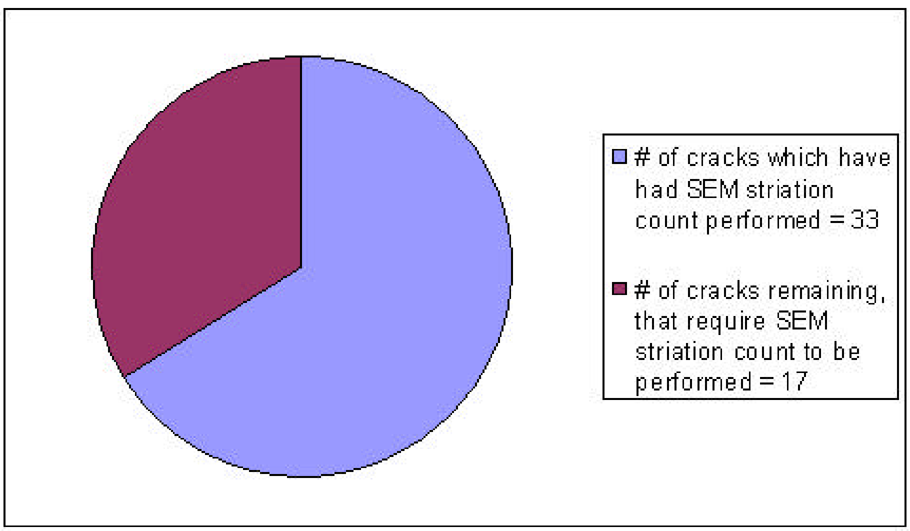


Figure 4: Damage Characterization Status as of 12/15/04

Teardown Matrix at DSG

DSG Teardown Matrix (Destructively examined with no testing)		
Panel Number	Initial MSD Condition	Description
Panel F1 FS 352 - FS 460 S-5L - S-6R	S-4L, FS 400-420: 4 cracked S-4L, FS 420-440: 5 cracked S-4R, FS 420-440: 1 cracked S-4R, FS 440-460: 1 cracked	Teardown only Includes UT tearstrap indications and lightning strike repair
Panel F2 (A and B) FS 850 - FS 969 S-5L - S-5R	None detected	Teardown only
Panel F3 FS 460- FS 600 S-2R to S-9R	FS 480-500: 2 cracked FS 500-520: 3 cracked, incl 1 from both sides FS 520-540: 8 cracked, incl 5 from both sides FS 540-560 12 cracked, incl 8 from both sides FS 560 - 580 6 cracked FS 580 - 600 10 cracked, incl. 3 from both sides	Right side, opposite FT1
Panel F4 FS 600- FS 720A; S-2R to S-9R	FS 600 – 620: 5 cracked, incl. 3 from both sides FS 620 – 640: 5 cracked, incl. 3 from both sides FS 660 – 680: 4 cracked, incl. 4 from both sides FS 680 – 700: 2 cracked, incl. 1 from both sides FS 700 – 720: 6 cracked FS 720 - 720A: 9 cracked, all from both sides	Right side, opposite FT2
Panel F5 FS 720A- FS 783 S-2R to S-9R	FS 720A - 720B 8 cracked, incl. 7 from both sides FS 720B - 720C 13 cracked, incl 11 from both sides FS 720C - 720D 4 cracked, incl. 3 from both sides FS 720D - 720E: 3 cracked	Right side, opposite FT3
Panel F6 FS 969-FS 1109 S-3L to S-9L	None detected	Left side, opposite FT4 Includes damaged stringer
Panel F9 FS 500 - FS 640 S-25L to S-27L	None detected	Teardown only

Task 10: Data Analysis

Contact Statement of Work

The contractor, in coordination with the FAA, shall analyze the crack data (patterns, distributions, sizes and shapes) calculated in Task 9 and demonstrate how to use this data to characterize MSD crack initiation, crack detection, crack link-up, residual strength and the WFD average behavior in the structures removed. Analysis methods will be developed to correlate the state of MSD at any point in time

a. Generate Stress Spectra: The contractor shall document the procedure to generate stress spectra representative of prior operation and usage for each structure removed. The basic aircraft usage (e.g. flight types, flight mix and flight hours actually flown) should be used in generating the spectra.

b. Crack Initiation/Initial Damage Scenario: The contractor shall determine and document how to estimate number of cycles to crack initiation and estimate the size, extent and distribution of cracks characterizing MSD initiation. Several methods are available including traditional empirical methods using extensive S-N test data with scatter factors, fracture mechanics-based Equivalent Initial Flaw Size (EIFS) concepts and a relatively new fatigue initiation model for lap joints, Eijkhout Model, outlined in National Aerospace Laboratory, NLR, report NLR-CR-2001-256. Use of test data in probabilistic analysis framework to determine crack initiation could be investigated. The contractor shall select one or more approaches and determine the applicability and feasibility in conducting WFD assessments.

c. Final Damage Scenario: The contractor shall determine and document the size, extent and distribution of MSD causing a reduction of residual strength below predefined levels defined in coordination with the FAA for the structure removed. Several approaches are available to estimate the final damage scenario including engineering estimates using sub critical conditions and new more rigorous techniques that were evaluated under FAA contract DTFA03-96-C-00027. The contractor shall select one or more approaches and determine the applicability and feasibility in conducting WFD assessments.

d. Conduct Crack Growth Analysis: Using the information developed in Tasks 3.9a through c above, the contractor shall conduct fatigue crack growth analysis for each structure removed. The analysis should be conducted from the initial damage scenario determined in Task 9b to the final damage scenario determined in Task 9c. Calculations should include the number of cycles to crack detection, crack link-up and to the final damage scenario. Government funded or other publicly available codes and methods should be used. Standard linear elastic fracture mechanics models or advanced crack closure-based fatigue crack growth models should be considered which are available in FASTRAN, NASGRO and AFGROW. The contractor shall select one or more approaches and determine the applicability and feasibility in conducting WFD assessments

e. Residual Strength: The contractor shall predict the fatigue crack growth and residual strength of the panels test in Task.7 per Task 9a –9d above. For the approach used the contractor will comment on the ability of the models to:

1. accurately analyze crack growth and residual strength (i.e. match empirical data to models)

2. accurately predict crack growth and residual strength (in the absence of empirical data)

<u>Contract Deliverables</u>

Phase 2: 10a: Data Analysis Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 10.

Phase 3: 10b: Data Analysis Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 10.

Phase 4: 10c: Data Analysis Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 10.

Phase 5: Data Analysis Report: 1 update reporting procedures, data, and results obtained in Task 10.

Data Analysis Goals for Phases 3 and 4

- Development of probabilistic model for global crack initiation
    - Bayesian learning capability
    - Distributions of initial flaws (stochastic initial damage scenarios)
    - Variability in stress states (residual and applied stresses)
    - Variability in crack distributions, directions, sizes and shapes
- Development of stress intensity solution for local crack state
    - "Tunneled" cracks
    - Multiple origins
    - Localized interactions
    - Changing crack front shapes (independent, amorphous, polynomial,)
- Development of stress intensity solution for lap joint cracking state
    - Crack arrays at each fastener hole (via two-step VCCT FEA iterations)
    - Local and global MSD interactions
    - Complex stress states (bending influence on crack shapes)
    - Curved and stiffened panel nonlinear affects
- Refinement of finite element model of fuselage skin lap joint bay
- Refinement of fuselage skin lap joint ABAQUS finite element 3-fastener "strip" model
- Completion 2D finite element modeling parametric study of riveting installation "defects"
- Development of 3D finite element modeling asymmetric models of riveting installation (academia)
- Explanation of fuselage lap joint inner skin lower row fastener hole crack pattern
        - Multiple cracking at lower quadrants around fastener holes

  - No cracking in upper quadrant around fastener holes

- Development of finite element models for superposition of applied stress state coupled with residual stresses from rivet models

- Explanation for crack initiation influence factors from damage characterization work

- Assessment and correlation of widespread fatigue damage test predictions

  - Crack distributions

  - Inspection Start Point (ISP)

  - Structural Modification Point (SMP)

  - Residual strength conservatism?

- Nondestructive Test Probability of Detection determination from damage characterization

- Nondestructive Test Probability of emerging inspection technologies evaluated

- Assess effectiveness of D-Box testing to simulate complex fuselage structural response

See the current Data Analysis report for more information.

Task 11: Inspection Capability Assessment

Contract Statement of Work

The contractor shall assess the capabilities the selected NDI used in this study to find and characterize damage. Results from the baseline NDI in Tasks 4 and 6 will be compared with the crack information obtained from destructive evaluations in Task 9 above. The contractor should document the limits of NDI detectability of small cracks.

Contract Deliverables

Phase 3: 11a: Inspection Capability Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 11.

Phase 4: 11b: Damage Characterization Report: 3 Quarterly updates reporting procedures, data, and results obtained in Task 11.

Goals from the Inspection Capability Assessment

- Assessment of the capabilities of selected NDI used in program
    - Conventional NDT methods in wide use by industry today
    - Emerging NDT technologies
- Parametric study of NDT effectiveness in detection of characterized damage
    - POD by crack location
    - POD by crack length
    - POD by crack direction
    - POD by crack shape
    - POD by number of MSD cracks per hole, bay, panel
    - POD by field inspection
    - POD by shop inspection
- Depository of all NDT related program activities

- Field inspections

- Shop inspections

- FAA Technician Level 1 Certification

- FASTER Test Inspections (update through 40,000 D-box pressure cycles)

Task 12: Documentation And Database Development

Contract Statement of Work

The contractor shall fully document all procedures, approaches, and data from the work outlined in Tasks 1 through 11 above in an electronic database. Information to be included in the documentation and database include, but are not limited to the following:

a. Rational for selection of the aircraft and structure analyzed

b. Procedures and data from field and pre-teardown inspections

c. Procedures used to remove structure from the aircraft

d. Procedures and approach used in the extended fatigue cycling and residual strength test using the FASTER facility

e. Data and results of all inspections including delivery of signal response data in the form of an electronic database

f. Data characterizing the state-of-damage including:

g. Fatigue crack distributions, locations, shapes and sizes

h. Damage initiation mechanisms and locations

i. Reconstructed fatigue crack growth histories

j. Quantification of corrosion, disbonds, fretting damage at faying surfaces, and other damage

k. Descriptions of the crack growth analysis methodologies used

l. Results of application of the methodologies as a means to analyze crack growth

m. Results of application of the methodologies as a means to predict crack growth

n. Description of the methods used to determine the MSD initiation, crack detection and crack link-up

o. Results of the analysis to determine MSD initiation, crack detection and crack link-up

p. Procedure and data from material characterization

q. Conclusions and recommendations specific to determination of MSD initiation, crack detection and crack link-up

r. General results and conclusions

Contract Deliverables

Phase 1: 12a: Initial Database: Develop structure and format of initial database using all the data used, collected, and developed Phase 1 of this contract. Databases shall be developed using commercially available software. Databases delivered shall also include documentation on database structures, data record specifications, their usage and data retrieval. The contractor shall submit the initial databases developed under this contract to the FAA 30 working days before the end of Phase 1 for review and comment. The FAA will have 15 working days to review and comment. The contractor shall then have 15 working days to incorporate comments provided by the FAA that are reasonable and within the scope of the work and then provide the FAA with the finalized databases in electronic format compatible with both a PC and commercially available software.

Phase 4: 12b: Draft Final Report: Five hard copies and an electronic version of the draft final report to the FAA at end of last quarter before of the contract performance period for review and comment. The contractor shall summarizing all work done under this contract in Tasks 1 through 12 and shall be delivered by the contractor to the FAA. The draft Technical Report shall include a description of the teardown process, visual inspection and NDI procedures, fractographic analyses results, data analysis results (EIFS, crack initiation and growth, etc.), assumptions made, methodologies applied with detailed theoretical derivations and computational formulations, models implemented (geometric configuration, material parameters, boundary conditions, etc.), and recommendations. The draft final technical report shall be prepared in standard electronic format and in accordance with the FAA Order 1700.8D, "Standards for Preparing, Printing, and Distributing Federal Aviation Administration Formal Technical Report."

12c: Final Report: The FAA will have 60 working days to review and comment on the draft final technical report. The contractor shall then have 30 working days to finalize the report with incorporation of comments provided by the FAA and provide the FAA with five copies of the finalized report along with all the electronic files in a format compatible with Microsoft Office version current at the time of the delivery.

12d: Final Database: Databases with all the data used, collected, and developed under this contract. Databases shall be developed using commercially available software. Databases delivered shall also include documentation on database structures, data record specifications, their usage and data retrieval. The contractor shall submit all databases developed under this contract to the FAA 60 working days before the end of the contract performance period for review and comment. The FAA will have 30 working days to review and comment. The contractor shall then have 30 working days to incorporate comments provided by the FAA that are reasonable and within the scope of the work and then provide the FAA with the finalized databases in electronic format compatible with both a PC and commercially available software.

Phase 5 During Q8, provisions were made to extend the deliverable schedule to accommodate the time required to complete FASTER testing of FT3 and FT4. At the end of Phase 5, an updated Final Report and Database will be issued to incorporate FT3 and FT4 data.

Goals for the Database:

The electronic database will contain all of the technical data delivered during the course of this contract.

- Tabular data such as crack lengths, inspection findings, rivet parameters, crack growth rates, etc. will be within fields in a table form.
- Graphical data such as fractography and NDT signal responses will be stored a graphic files (e.g., *.jpg) linked to caption tables
- Report data such as procedures, results, and conclusions will be *.pdf files linked to user input forms

The Database will be the ideal tool to distribute these results to the industry

- Database will be the tool that institutionalizes the teardown and evaluation process
- Standard queries and reports will filter data as needed

- Expertise in MS Access will not be required for use, but knowledgeable MS Access users will be able to design very sophisticated custom data queries
- Database should be contained on a data CD or CD collection.
- Database must be packaged such that it is easily transferable to the FAA or industry.

Standard Format Codes

Two standard format codes have been developed as database keys. The Longcode is a unique, intuitive standard format for each finding, based on its location on the airplane. In contrast, a typical numbering system is one where structures, cracks, strain gauges, etc. are numbered sequentially. This method is simple, but not intuitive. For a typical system, a complex graphical map is needed to correlate each item to its location. This works well for a reasonable number of details, but is untenable for a large number of details. This research program will database information on more than 12,000 holes and tearstrap locations, and the format must be readily applicable to any new findings.

The Shortcode is an abbreviated format that still intuitive, but is short enough to inscribe on fractographic specimens or conveniently list in tables. The short code is used as the unique key to link records in the database. See the Initial Database Report for additional information.

APPENDIX A: FULL-SCALE TEST AND TEARDOWN LITERATURE REVIEW

A Review of Published Literature on Aircraft Full-Scale Fatigue Testing, Destructive Evaluation and Teardown

BACKGROUND

Modern aircraft structures are designed using a damage tolerance philosophy. This design philosophy envisions sufficient strength and structural integrity of the aircraft to sustain major damage and to avoid catastrophic failure. High replacement costs and competition have led airline operators to use their airplanes beyond the original design service objective. The structural aging of the aircraft may significantly reduce the strength below an acceptable level, an important safety issue. Fatigue is a major economic and safety problem for aircraft based on historical evidence. Aircraft fatigue problems were studied since the late Fifties. In the USAF, the Aircraft Structural Integrity Program was initiated in 1958.

An aging fleet has a higher probability of fatigue-initiated cracking and eventually results in widespread fatigue damage (WFD). WFD is defined as the simultaneous presence of fatigue cracks at multiple structural details that are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement (maintaining the required residual strength after partial failure). There are two types of WFD: multiple site damage (MSD, fatigue cracks in the same structural element) and multiple element damage (MED, fatigue cracks in adjacent structural elements).

According to Goranson,[1] the following aircraft structures are susceptible to WFD:

Fuselage	**Wing and empennage**
longitudinal skin joints, frames, and tear straps	chordwise splices
circumferential joints and stingers	rib to stiffener attachments
frames	skin runouts of large doublers
aft pressure dome outer ring and dome web splices	stinger runouts at tank end ribs
other pressure bulkhead attachment to skin and web attachment to stiffener and pressure decks	
stringer to frame attachments	
window surrounding structures	
overwing fuselage attachments	
latches and hinges of nonplug doors	
skin at runouts of large doublers	

The in-flight structural failure of an Aloha Airlines Boeing 737 on April 28, 1988 reinforced the aging aircraft study. The failure precipitated from the link-up of small fatigue cracks extending from adjacent rivet holes in a fuselage lap-splice joint. This caused approximately 18 feet of the upper crown skin and structure to separate from the fuselage. The Aloha Airlines accident created a revolution in the aircraft community. The problems associated with aging aircraft have to be quantified and the methodology to ensure the structural integrity of airplanes has to be reassessed. Federal Aviation Administration (FAA), National Aeronautics and Space Administration (NASA), and the Department of Defense (DoD) have sponsored research on the development of various analytical methodologies to investigate aging aircraft problems. These analyses are performed in accordance with the airworthiness requirements FAR/JAR 25.571. As part of these efforts, the Full-Scale Aircraft Structural Test Evaluation and

Research (FASTER) facility was established.. According to Bakuckas and Carter, full-scale fatigue testing will provide data to enable calibration and validation of prediction methodologies and will serve as a test bed to evaluate the sensitivity and effectiveness of standard and emerging inspection technologies to detect small cracks. The data generated from this effort will be used to calibrate and validate WFD assessment methods. The analysis methodology would allow engineers to maintain the aging fleet economically while insuring continuous airworthiness. The experience and knowledge gained from this project will enable the FAA to issue essential rules, policy, and advisory circulars pertaining to the prevention of WFD. The findings from full-scale fatigue tests will help to determine service inspection intervals, quantitatively evaluate inspection findings, and allow the safe operation of the current aging fleet beyond the original design service goal.

The main objective of this work is to conduct a literature review of the aircraft full scale fatigue testing and teardown approaches and methodologies that have been utilized for aircraft components and structures, inspection methods, and damage mechanics and models for life prediction.

FULL SCALE FATIGUE TEST

In the early years most of the full scale fatigue tests performed were conducted to evaluate the adequacy of a specific proposed design or to check the weaknesses in existing design.[2,3] Since the 1988 Aloha Airline 737 aircraft accident, full-scale fatigue tests has been motivated by the desire for safe operation of the current aging fleet beyond the original design service objective.

There are many factors to be considered in the full-scale fatigue testing. The following topics are covered in this section:

- Full-scale fatigue testing facility

- Fatigue characteristics such as fatigue life, crack initiation, WFD distribution, WFD linkup, and the effects of multiple-site cracking on the residual strength of fuselage structures.

Full-Scale Fatigue Testing Facilities

FAA developed a Full Scale Aircraft Structural Test Evaluation and Research (FASTER) facility to apply realistic flight load conditions to large, full-scale, curved sections of fuselage structure.[4,5,6,7,8,9,10,11,12,13] This facility test curved panel sectors. It is more suitable to simulate typical aircraft pressure vessel structure at the crack tip than the flat panel testing method. Details of this test capability are provided in Bakuckas report.

Four types of fuselage panels were studied using FASTER at FAA: (1) two configurations with longitudinal lap splices, (2) two configurations with circumferential butt joints. For each configuration, one contained only a lead crack and the other contained a lead crack with multiple cracks. Fatigue crack formation and growth was monitored and recorded in real time using the Remote Controlled Crack Monitoring (RCCM) system. A Self-Nulling Rotating Eddy-Current probe system developed by NASA Langley Research Center is also used as crack inspection tool. According to Bakuckas and Carter, critical fatigue areas include FS 360-FS 720, FS 1009-FS 1130, and S 4R-S 4L. The chemical composition of the material for skin, frame, stringer, shear clip, and fastener, etc. can be identified from drawings and analysis results from either x-ray florescence spectroscopy (XRF) or energy dispersive spectroscopy (EDS). Reported test length is from 16,000 to 20,800 cycles. The test panel dimensions are shown in Table 1.

Table 1: Test panel dimensions.

LONGITUDINAL	**CIRCUMFERENTIAL**	**RADIUS**
120”	68”	66”
7 stringers with a 7.5” spacing	6 frames with a 19” spacing	

Table 2: Applied Loads.

Test Type	Load Type	Maximum Load			
		Pressure (psi)	Hoop (lb/in)	Frame (lb/in)	Long (lb/in)
Strain Survey	Quasi-Static	16.0	878.6	177.4	0
Strain Survey	Quasi-Static	0	0	0	528.0
Strain Survey	Quasi-Static	16.0	878.6	177.4	528.0
Fatigue	Cyclic	16.0	878.6	177.4	528.0

Sponsored by FAA, Foster-Miller, Inc[14, 15] also conducted full scale fatigue tests of fuselage panels.

Fatigue test panel characteristics are shown in the Table 3.

Table 3:Test panel configuration:

Panel length (inches)	120
Panel width (inches)	68
Panel radius (inches)	75
Number of frames	6
Number of tear straps	11
Number of stringers	6
Frame spacing (inches)	20
Tear strap spacing (inches)	10
Stringer spacing (inches)	9.6
Skin thickness (inches)	0.036
Tear strap thickness (inches)	0.036
Skin and Tear strap material	2024-T3 Al alloy (clad)

The pressure for test panels was between 1.0 and 9.5 psi. The rate of loading was 0.2Hz or 720 cycles per hour. Study focused on the distributions of cracks along the upper rivet row at different points in time as the panel was cycled. The tested fatigue life for a lap splice panel loaded to a pressure differential of 8.5 psi is approximately 75,000 cycles. The first linkup of multiple cracks occurred at about 92% of this life.

This panel deviates in some design features from an actual commercial aircraft. A tear strap and filler strip combination was used in place of the waffle doubler design in actual aircraft. Larger universal head rivets are used instead of countersunk rivets.

Boeing full-scale fatigue testing facility was introduced by Miller and Gruber et al.[16, 17, 18, 19, 20, 21] There are two types of barrel pressure test fixtures at Boeing: wide-body and narrow-body. A fixture with a radius of curvature of 74 inches is applied to match Boeing's typical narrow body aircraft (727, 737, and 757), and a fixture with a radius of curvature of 127 inch for wide-body aircraft (747, 767, and 777). Both fixtures are 20 feet in length. The overall geometry of the fixture is consistent with typical fuselage design: 7075-T6 frames at 20 inch (508mmm) pitch and 2024-T3 clad stringers at a 9.25 inch (235"). Each fixture has a 10 feet by 10 feet rectangular cutout, designed to install the test panels.

Air is applied for simulation the true dynamic conditions in a fuselage structure. Greater air compressor capacity would help reduce cycling time. Compressed air is used as the pressurizing medium. Differential pressure is 90 psi produced by 5 electric compressors. Air flow is regulated by a digital valve. Cyclic rates are 1 min per cycle in the narrow -body fixture. 0.1 inch thick rubber air bladders are used to reduce air leakage.

The data acquisition system consists of up to 300 channels for recording test information, principally from strain gages. Most tests are remotely monitored using video cameras. The resulting film is used for review and analysis. A test crack is introduced by inserting a 5-inch sawcut at the test location.

Being conducted full-scale fatigue and teardown studies for aging aircraft. But there is no data released on EIFS determined from aging aircraft.

Using air as pressurization medium in full-size structure raises significant safety issues. The number of tests that are required to generate the necessary data would be both expensive and time-consuming due to the size of the full-scale facility.

Lockheed Martin Aeronautics Company established a service life assessment program in the 1990s.[22, 23, 24] Full-scale fatigue test was performed for P-3C. The test duration goal is to reach two times service life and testing may be continued at 10,000 fatigue test spectrum hour. The areas selected for fatigue testing include: (1) center wing-lower surface, (2) outer wing-lower surface, (3) outer wing-upper surface, (3) main landing gear, (4)nose landing gear, (5) nacelle, (6) fuselage, (7) empennage, (8)control surfaces, (9) additional center wing section locations, (10) additional vertical stabilizer locations. Miller et al reported a methodology for development of the loading spectra for the wing/fuselage and empennage test for the S-3B Full-Scale Fatigue Test program. There is limited information in the literature on details of the test facility.

NASA Langley Research Center[25,26] and FAA jointly designed and conducted fracture tests on 40-inch wide sheets made of 0.063-inch thick 2024-T3 Al alloy with and without stiffener. Stiffeners are made of 0.087-inch thick 7075-T6 Al alloy. The central stiffeners were cut along the crack line. Open holes were machined into the sheet at the required rivet spacing along the crack line but rivets were not installed. Five different crack configurations were tested: A single center crack, a single center crack with an array of 12 holes on either side of the lead crack, and a single center crack with three different equal MSD crack lengths (0.01, 0.03 and 0.051-inch) at the edge of each hole.

United States Air Force (USAF) established Aircraft Structural Integrity Program (ASIP) in 1958. The goal of this program is to control structural failure of operational aircraft, determine methods of accurately predicting aircraft service life, and provide a design and test approach that will avoid structural fatigue problems. MIL-A-8867B (was given later in AFGS-87221A) states that full scale durability test should be run for a minimum of two lifetimes unless the economic reached prior to two lifetimes. It is still not certain if all the potentially safety problem areas can be determined based on this type of test.

According to Lincoln[27], several references established the duration and/or the severity of the full scale testing. Using an average spectrum is not economical because of the testing period is too long. The practical way to complete the test in a timely manner is to increase the severity of the test spectrum.

It is necessary to conduct a fractographic examination if the crack growth is faster than predicted. Local stresses and fracture data for the material used in the full-scale test article should also be determined in this case. An assessment should also be made to determine the implication on the damage tolerance derived inspection program

Airbus also conducted barrel full-scale fatigue tests.[28,29,30,31,32] Schmidt and Nielsen [30] presented information on the development of load spectra for center fuselage and wing test and the rear fuselage test for medium A330 and long range A340. The typical A330 mission contains an average block time of 90 minutes and an operation altitude of 35,000 ft. Two typical A340 missions are a short mission with 75 min./35,000 ft and a medium range mission with 405 min./39,000 ft.

According to Boetsch and Beaufils, the loading of the specimen is carries out through the following devices on the fuselage: loading jacks, loading trees of the cabin floor and the pressurization is achieved through the test jig. The ground loads are applied to the nose landing gear by three jacks. The areas chosen for full scale fatigue tests on the Airbus A320 are as follows:

- Forward fuselage
- Wing/center fuselage
- Rear fuselage
- Horizontal stabilizer
- Vertical stabilizer

The test is scheduled to be 120,000 simulated flights (2.5 lives with fatigue life design of 48000 flights). The load spectrum follows "flight by flight" procedure. For the forward fuselage section, the test load include (1) take off, landing, taxiing, (2) air loads: gust and maneuver, (3) cabin pressure.

According to Schmidt, Airbus full scale fatigue test simulated a minimum of two life times, and was multi-section testing, such as (1) front fuselage, (2) center fuselage/wing, (3) horizontal tail, and (4) rear fuselage.

National Laboratory NLR the Netherlands has a fuselage panel full-scale fatigue test facility.[33] The fatigue test loads simulate cabin pressurization in radial and axial direction and axial loads representative of fuselage bending. The performance of the test set-up is evaluated. The test results indicate the suitability of the biaxial load introduction systems to load a panel comparable to a panel in a pressurized fuselage subjected to bending. Test duration is 180,000 flights. The chamber pressure are reproduced within 3 percent.

National Aerospace Laboratory (NAL) in Japan[34] has a barrel testing facility. A pressurized fuselage section of 2.5 meters long, 1.25 meters diameter can be tested. The barrel structure basically consists of four skin panels stiffened by Z-section stringers and attached to four Z-section ring frames to make up five-bay barrel structure. Four panels overlap and join each other by three rivet rows to form four longitudinal lap splice joints. There are only 2.5 bays with nominal skin thickness 0.81mm. Both ends of barrel specimen are reinforced by doublers to prevent undesirable failure. About 200 strain channels monitoring 2- and 3-element rosettes were located within third bay mostly on the outside surface of skin. Whenever possible, 2 or more gages were used to average data over similar location. Deformation of skin, lap joints, and structural details of the central bay was monitored by approximately 100 strain gages, displacement dial gages, and extenzometer.

Department of Defense, Defense Science and Technology Organization (DSTO), Aeronautical Research Laboratory, Australia, designed and built loading rigs that can perform full-scale fatigue testing for aircraft wings and test them to destruction (refer to http://www.dsto.defence.gov.au/index.html). Some information on the testing equipment was provided by Payne [2].

Structures, Materials and Propulsion Laboratory, Institute for Aerospace Research (IAR), National Research Council Canada also conducted full-scale fatigue testing (refer to http://www.nrc.ca/iar/structures_1a.html). A large test bay is equipped with computer-controlled mechanical and hydraulic loading devices and sophisticated data acquisition systems.

A major full-scale fatigue test of an F-18 wing is currently in progress. Loads are applied via 63 computer-controlled servohydraulic actuators while data from over 600 strain gauges are acquired and monitored automatically.

Lessons learned: Full-scale fatigue tests of aircraft structures / components that have not been in service may not give accurate information on crack initiation because (1) full-scale fatigue test results may not be representative of the actual operational load spectrum, and (2) the potential influence of environmental exposure may have on the crack initiation process is neglected. Most of the tests (except one project conducted at Japan) do not incorporate corrosion and/or accidental damage that can accelerate fatigue cracking. The real operation may deviate from the typical full-scale profiles the fatigue and damage tolerance behavior of testing results may deviate from the in-service aircraft. According to Hewitt[35], full-scale aircraft structural fatigue tests are extremely complex from a control systems view point. There are usually a large number of actuators with significant interactions between them. Control is made more difficult because load cells usually move with the actuators.

Fatigue Characteristics

<u>Findings from FAA</u>

In 2000, Akpan et al reported that symmetric, collinear crack propagation under constant-amplitude fatigue loading was observed for the two panels tested. The first panel had a longitudinal lap splice, and the second panel had a circumferential butt joint. Reasonable agreement was obtained between experimental fatigue crack growth data and predictions relying on the Mode I stress-intensity factors calculated using finite element analyses of the test panels.

Residual strength tests were conducted on each panel after the fatigue loading. For the curved panel with the longitudinal lap splice, the initial damage consisted of a two-bay crack with a length of 25” with the central frame cut, and for the curved panel with the circumferential butt joint, the initial damage consisted of a four-bay crack with a length of 19” with the central stringer cut. Failure of the panels occurred at 11.14 psi and 17.5 psi, respectively, well above typical operational pressures of 8 psi for narrow-body aircraft. Thus, these results show that curved panels representative of narrow-body aircraft fuselage structure with large cracks and broken frames and stringers are capable of withstanding loads in excess of typical in-flight operational pressures.

Ahmed et al performed geometric nonlinear finite element analysis to predict the strain distributions and stress in the panel. FEA analysis results and testing results are found in good agreement. It was concluded the analysis assumptions and boundary conditions were valid. Other parameters governing fatigue crack initiation and growth of the curved panels were not present yet.

Findings from Boeing

Gopinath reported that most of the local cracks were initially detected past 30,000 full pressure cycles for 747-100 test article. The cracks located at mid-bay between frames, initiating out of discrete upper row fastener holes. At 40,000 full pressure cycles, none of the cracks detected during cycling, with the exception of one location, had linked up between fasteners. No cracks were detected in the aft fuselage skin splices. The majority of cracks detected during cycling on the internal fuselage structure were on

- frames
- passenger entry door cutout structure
- passenger entry doors
- upper deck floor beams

A significant percentage of these cracks were detected on the internal structure in the nose section. Cracks on the internal structure of the nose section were reported in the fleet on airplanes with 10,000 to 17,000 full pressure cycles. The majority of the cracks on the internal structure of the other fuselage section were detected after 32,000 full pressure cycles.

Some local cracking was detected on skin lap splices after 33,500 full pressure cycles in the nose section of the 747-400 forward fuselage. The initial crack lengths at detection by high frequency eddy current inspections were 0.05 to 0.07 inches long. The cracks were generally at mid-bay between frames, initiating out of discrete upper row fasteners. The maximum crack length at 40,000 cycles was 0.20 inches. None of the cracks linked up between fasteners at 40,000 full pressure cycles. Very few cracks were detected in the rest of the fuselage structure. The two significant findings were cracks in the upper deck stairwell cutout, and skin cracks in the electrical equipment cooling cutout unique to the -400 model. These cracks were detected past 33,500 test cycles.

Lessons learned: This research is lacking major technical issues: (1) no inspections conducted on fuselage lap joint inner skin lower fastener row. (2) residual strength conclusion did not consider MSD and accidental damage (large damage capability).

According to Goranson et al,[18, 19, 20] the 737 aftbody section test extended the pressure cycling from 59,000 cycles at retirement (18 years of service) to 130,000 total cycles. At 79,000 total cycles, corresponding to 24 years of service, MSD of approximately 0.09-in length beyond the fastener head was visually detected in the skin at seven upper row countersunk fasteners along Stringer 4R. The 737 test revealed no major new economic- or safety-related fatigue findings. One of the significant findings was cracking at typical frame splices below the entry door. Approximately one-third of the fasteners in the critical row developed cracks. The damage detection period prior to linkup (during which safety inspections are carried out) ranges from 10,000 to 40,000 flight cycles, depending on lap joint condition (corrosion, disbond). Following crack linkup, a major crack reached a longitudinal length of 32 inches with three tear straps failed before a change in crack direction caused “skin flapping” and safe decompression at 100,600 total cycles equivalent to about 30 years of typical service usage (7 years without repairs since initial crack detection). Repairs of the aft pressure dome permit continued testing of the section to an equivalent of 130,000 total pressure cycles (40 years of operator service). The 747 test extended the pressure cycling from 20,000 service cycles to 40,000 total cycles.

Test sections are modeled using finite element techniques, and analysis results are compared with a comprehensive set of strain gauge and crack opening displacement measurements. The structural modeling of crack behavior can be refined , as necessary, to provide a validated tool to determine the effect of crack location and trajectory and material nonlinearity on the residual strength of the test panels.

Piascik et al [36] conducted full scale testing of fuselage structure to demonstrate structural airworthiness to 60,000 full pressure (0 to 9.0 psi) cycles, i.e. three times the minimum economic design objective. Cracks in the fuselage skin lap splice joint were observed in a few localized regions of the structure. Fatigue crack initiation occurs in regions of high fracture toughness located at or near rivet hole corners, surface discontinuities (burr, dents, etc.) and abraded (fretted) surfaces. Fretting fatigue damage black oxide debris, was noted at every rivet hole/lap splice interface. Most fretting initiated fatigue cracks were contained in the upper row; here, all fatigue cracks initiated in the outer skin at the inner/outer skin interface in regions that exhibited increased surface abrasion (fretting damage).

Gruber et al found that the crack growth rate of the leading crack in the lap joint was greater in the presence of simulated MSD. Without MSD the lead crack arrested in each fastener hole, and additional cycles were required to extend the crack out of the opposite side. For the floating frame panel geometry (no shear ties) and crack lengths tested in this program, the presence of 0.05-inch MSD in the lap joint upper rivet row reduced the residual strength of the panel by 20% compared to panel containing no MSD. During the residual strength portion of the tests, the tear straps bridging the crack tips of the two-bay lead crack remained intact until the final failure load was reached. However, the test results are inconclusive as to whether tear strap failure led to dynamic extension of the lead crack.

Meyer et al [37] tested 2,400 groups of fatigue test data to determine if the fatigue failures were a function of any particular variables, e.g., specimen configuration, alloy, load type, etc. The only variable that consistently affected fatigue scatter was the type of loading, i.e., spectrum or constant amplitude. Constant amplitude testing typically resulted in more scatter compared to spectrum testing. The differences are attributed to the fact that the spectrum test results were effectively low cycle fatigue due to the frequent occurrence of high loads while the constant amplitude results were from high cycle fatigue where more scatter exist. These full-scale fatigue tests are used to estimate the projected fatigue life distribution.

Findings from NASA Langley Research Center

NASA Langley research laboratory conducted the Airframe Structural Integrity Program in support of the aging commercial fleet.[38,39,40,41,42,43,44] Newman et al did a lot of work on the aging aircraft fatigue investigation. He concluded that crack nucleation is associated with cyclic slip and is controlled by the local stress and strain concentration. Micro-crack growth is the growth of cracks from inclusions, voids, or slip bands, in the range of 1 to 20 μm in length.

Fatigue cracks were present at virtually every rivet hole in the top row of rivets. The cracks ranged in size from about 50 μm to several centimeters. Crack initiation mechanisms included high local stresses, fretting along mating surfaces, and manufacturing defects created during the riveting process. The cracking behavior in each bay was similar and the results of the fatigue marker bands were relatively independent of rivet hole location.

Harris stated that MSD would require stress intensity factor solutions for three fundamentally different levels of crack sizes. For very small cracks below the damage tolerance regime, the finite element method will be used to generate solutions to three-dimensional crack configurations such as surface and corner cracks initiating at countersunk rivet holes. The results from FEA modeling can be used for the stress intensity factors for several basic crack configurations. After the cracks extend through the wall thickness and beyond the rivet head, the cracks will be in the detectable range and amenable to the damage tolerance philosophy.

The two-dimensional boundary element method will be used to generate stress intensity factors for MSD crack configurations prior to extensive link-up. After MSD crack link-up, the cracks are quite large and crack growth will be rapid. The stress intensity factor for these cracks will be strongly influenced by the geometric nonlinear response of the stiffened fuselage shell structure. The integrated fracture mechanics and fuselage structure analysis methodology are verified by the curved panel and subscale barrel test program.

According to Kurth and Bigelow[45], the onset of WFD, as well as the structure's residual strength, are not deterministic quantities, rather there is some probability that either the onset of WFD has occurred or the residual strength has fallen to unacceptable levels.

Findings from Airbus

According to Boetsch and Beaufils, there were 160 fatigue damage findings at the end of full-scale fatigue tests of A320. The majority were found on secondary structures.

Lessons learned: No details of these fatigue damage were disclosed.

Santgerma et al conducted full scale fatigue test for widespread fatigue damage assessment in A300 circumferential joint structure. During full-scale fatigue test, MSD occurred and was detected in a circumferential joint situated in the forward fuselage section, ahead of the passenger door. This 3-rivet row joint is loaded mainly by cabin pressure. The damaged area was removed for fractographic analysis. Finite element methods and Monte-Carlo analysis was applied to study the fatigue behavior. The following structures are identified to be susceptible to multiple site damage:

- Longitudinal and circumferential fuselage joints
- Successive fuselage frames
- Wing/fuselage attachment

- Front pressure bulkhead
- Integrally stiffened panels at stringer run-outs

Lawford presented a summary of a test program used to support the development of widespread fatigue damage analysis methods for Airbus A300. The areas of the right hand wing that are susceptible to widespread fatigue damage for A300B2 are as follows:

- Rib 9 chordwise skin panel joint, top and bottom
- Bottom skin, stringer web runouts at tank end rib 14
- Rib to skin joints, top and bottom skins
- Ends of the top skin reinforcing doubler

WFD fatigue tests were conducted for (1) top skin chordwise panel joint at Rib 9, (2) bottom skin chordwise joint at Rib 9, (3) bottom skin, stringer web runouts at tank end rib 14. The test results are present in Table 4. Monte Carlo approach was used to define skin crack initiation periods.

Lesson's learned: No details of Monte Carlo approach are provided in this report

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Table 4: WFD tests results.

Panel	Target	Achieved	Comments
Rib 9 top skin joint	90,000	92,000	Load cycling stopped with MSD and MED present. Teardown inspection of panel has been completed. Total of 75 cracks found, 15 cracks>10 mm. Load factor applied to test loads=1.3 Equivalent aircraft cycles = 250,000
Rib 9 bottom skin joint	90,000	62,000	Load cycling stopped with MSD present in skin panel. Load factor applied to test loads=1.3 Equivalent aircraft cycles =180,000 Teardown inspection has been completed.
Rib 14 bottom skin, stringer web runout	90,000	84,000	WFD testing completed. MED initiation at approximately 68,000 FCs demonstrated - cracks in 2 adjacent stringers. Load factor applied to test loads = 1.3 Equivalent aircraft cycles = 240,000 Cracked stringers were repaired to aircraft standard and a further 16,000 FCs applied to verify repair performance and allow further WFD initiation.

Schmidt reported that full-scale fatigue tests of A300B2 revealed only a few cases of multiple site damage. In the described fuselage area only three cases of multiple site damage have been detected between 1.6 and 2.2 times of economic repair life. For these areas modifications and inspection programs have been defined beginning at 1/3 of time-to-detection with repeat inspection according to the crack propagation behavior and the capability of the defined NDT method. The effect of debonding due to corrosion on the fatigue life of the longitudinal joints has been investigated. The barrel test revealed a fatigue life of more than three times the economic repair life for an intact longitudinal joint. The deterioration was the result of increased bearing stress, reduced net section and changed eccentricity leading to a change of crack origin and crack shape up to a certain length.

Findings from Japan

According to Dybskiy et al, some outer skin cracks were found after 90,000 cycles, and a flapping failure happened after approximately 119,000 cycles for the first test. For the second test, cracks were found penetrated outer skin in a lap joint after 44,000 pressure cycles, and the test lasted 79,500 cycles.

Furuta et al[46] at the Kawasaki Heavy Industries (KHI) and NAL tested lap-joint specimens with two- or three-rivet rows using 100-degree countersunk rivets. These tests were conducted under either ambient conditions or immersed in a 3.5% salt-water solution. Harris et al analyzed these test data, refer to [41]. Results show that the fatigue-life of testing panels exposed to salt water is reduced by a factor of about ½ or 1/3 compared to the fatigue-life in ambient laboratory air environment. Cracks initiated and grew at several rivet holes located in the midbay of the panel, linked up at about 10 mm half-crack lengths, and caused panel failure.

Findings from IAR Canada

Feastaugh[47] et al reviewed loading and fatigue characteristics of longitudinal fuselage splice. The primary fatigue load on a longitudinal fuselage splice is fuselage pressurization to an approximately constant amplitude of around 8 psi per flight. During a lifetime of 20 years, a long-range transport aircraft might be subjected to 20,000 pressurization cycles, while a medium range aircraft might be subjected to 80,000 pressurization cycles. This pressurization subjects a splice to a combination of hoop tension, longitudinal tension, and out-of-plane bending. The out-of-plane bending is associated with two effects. The first is pillowing of the skin between stiffeners. The second is the distortion, known as secondary bending, caused by the action of hoop tension on the eccentricity inherent in single-shear lap or butt splices.

According to authors, the general characteristics of crack growth in fuselage appear to be as follows:

1. Under the rivet heads, crack grows much slower than the growth beyond the rivet heads. In 737 splices, the hidden growth period is similar in magnitude to the visible growth period to first linkup.

2. Clusters of cracks develop in one or more frame-bays, away from frames and straps. The cracks may stretch across most of the width of a frame bay, or they may consist of only 2 or 3 cracked rivet holes.

3. Adjacent cracks will link up to form a lead crack eventually. The lead crack grows faster than unlinked cracks, and tends to dominate subsequent crack within a frame-bay.

4. In most cases, cracks grow longitudinally, along the rivet row. In some cases cracks grow obliquely, indicating the presence of in-plane shear stress. In either case, cracks that have overlapped generally curve towards and eventually intersect the opposing crack.

5. The combination of a large MSD cluster and uniform crack lengths result in a relatively short period between the first link-up and the development of linked crack across the full frame-bay.

6. Small cracks and non-uniform cracks will need a relatively long growth period after first link-up. On the 747 the growth period from first link-up to critical length varies between 1,000 to 10,000 cycles.

7. A lead crack grows rapidly until it reaches critical length, which is typically one to two frame bays for a single crack. Frames and straps can slow down a linked lead crack.

8. If there are cracks in the next frame-bay, the lead crack may grow faster and its critical length may be shorter. If the intervening frame is still intact, the crack may have to be close for these interaction effects to be significant

9. A lead crack may turn in a circumferential direction and form a skin flap in narrow-body aircraft. The flap may initiate before or after the crack has become unstable, and may occur at

either the first or second frame encountered. The change in crack direction may occur abruptly, close to the frame, or it may be gradual. Narrow-body fuselages are designed to promote this type of fail-safe crack behavior, but it is less likely to occur if the crack is in a splice, particularly if there are cracks ahead of the lead crack.

Findings from NLR the Netherlands

Full-scale fatigue studies have been conducted at NLR.[48, 49,50,51] Multiple site damage fatigue cracks initiate at faying surfaces near or at rivet hole corners. MSD fatigue initiation lives were significant fractions of the total lives.

Estimates of MSD fatigue initiation lives are shown in Table 5.

Table 5:. MSD fatigue initiation.

Aircraft Type	MSD rivet row	Flights or simulated flights	
		Flights to first crack " initiation"	Total flights
BAC 1-11	outer sheet upper row inner sheet, lower row, port	50,000 56,000 47,000	75,158
F100	outer sheet upper row inner sheet lower row	60,000 70,000	126,250
B747-400	outer sheet upper row	5000-15,000	60,000

Wanhill et al reported that aircraft structures are susceptible to fatigue and corrosion damage, notably at joints. Interaction between fatigue and corrosion is possible as aircraft age. Longitudinal lap splices from several types of transport aircraft pressure cabin were dissembled and investigated for multiple site damage fatigue cracking and corrosion. It was found that pressure cabin lap splices have either a fatigue problem or a corrosion problem. It is concluded in this report that MSD fatigue cracking was not initiated by corrosion. Fatigue fractography examination found that most MSD-susceptible rivet row was the upper one in the outer sheet of each lap splice. However, other rivet rows were also susceptible, such as the lower one in the inner sheet. MSD fatigue crack initiated from the faying surfaces near or at the rivet hole corners. Surface conditions of the materials such as cladding, anodizing, priming, sealant, adhesive bonding are very important. Fretting may play an important role for crack initiation and early crack growth.

Table 6:MSD behavior findings from NLR.

Aircraft	Findings
F28-4000 Service Aircraft	The total lengths of MSD for two aircraft measured were 530 mm and 330 mm. The cracks initiated at many sites along the faying surface edges of the outer sheet dimpling cones. The cracks were mechanically induced with no corrosion evidence. The large number of initiation sites suggests that fatigue cracking began soon after aircraft entered service.?
BAC 1-11 service aircraft	MSD was along about 500 mm of port and starboard lap splices. The cracks initiated at a variety of locations, though mainly at faying surfaces close to the rivet holes. No corrosion initiation signs.
F100 full scale test (indoors)	MSD extended over several frame bays having poor adhesive bonds. The cracks initiated from the faying surfaces, mostly at multiple sites close to the rivet holes. No corrosion evidence.
B 747-400 full scale test (outdoors)	MSD extended several frame bays. The cracks initiated at a variety of locations, though many at faying surfaces and rivet hole/faying surface corners. There was no evidence of corrosion, even though the test was outdoors.

Secondary fatigue in the B727-100 lap splice initiated from intergranular corrosion cracking (exfoliation corrosion and stress corrosion). 12 small fatigue crack were found at six rivet hole positions, without any evidence of fatigue initiation due to local corrosion (pitting). Ten of the cracks occurred inside the rivet holes in the inner sheet. This behavior is distinct from MSD, where the majority of cracks initiated from faying surfaces.

Some of the beach marks observed were attributed to periodic changes in local environment. It also suggested that modeling and prediction of early MSD crack growth may need not account for environmental effects beyond that of normal air. But the tests performed are not at the same frequency as in-service pressure cycling, and the crack growth rate is not similar to those early MSD in the lap splices.

Wanhill and Koolloos investigated multiple site damage (MSD) fatigue and corrosion longitudinal lap splices from several types of transport aircraft pressure cabins. The lap splices were from three Fokker F28s, a British Aerospace BAC 1-11 and a Boeing B727-100 and a full-scale test on a Fokker F100. They concluded that corrosion pitting is not responsible for initiating MSD fatigue cracks in lap splices using 2024-T3 Alclad skins. Different early MSD fatigue crack growth rates obtained fractographically from different sources. The data show the following important features:

- All crack growth rates, from cracks 30 μm - 5 mm in size, were above 10^{-8}m/cycle
- The transverse (through-thickness) crack growth rates were, on average, fairly constant and remarkably similar.
- The longitudinal crack growth rates were also remarkably similar, considering the different aircraft types and lap splice configurations.

They also reported that a DAIS imaging detected severe corrosion better than eddy current.

In their work, the relationship between MSD fatigue failure and corrosion was not identified.

TEARDOWN

For determining the crack initiation, onset of WFD, validation of NDI/NDE techniques, the teardown inspections of actual fleet aircraft provide the most reliable method [Bakuckas and Carter, Boetsch and Beaufils, Coulter[52], and Cochran[53]]. Tear down investigations using full scale fatigue specimens result in indispensable knowledge for structural mechanics as well as for metrology and inspection technology. When full-scale fatigue testing is completed, the components will be disassembled and inspected to identify the location of all defects. Critical areas are then examined by breaking components apart in places where cracks might be expected. All significant cracks are then subject to 'fractographic' analysis which determines crack growth rates. The data for new significant locations is used for forming management strategies. The knowledge gained from the tear down investigation on the actual damage is a key element for the development of crack initiation modeling, onset of WFD, and development of reliable NDI procedures.

The following can be expected from teardown:

- Determination of the condition of the test specimens subsequent to completion of the fatigue test.
- Visual and NDI/NDE inspection fatigue samples in assembled and disassembled condition.
- Fractographic and metallurgical examination of critical cracks after visual and NDI inspection. Recording the fractography. Measuring the crack size.
- Calibration and correlation crack initiation models such as Equivalent Initial Flaw Size model, Eijkhout Model, and fatigue corrosion model.
- Validation and correlation of the NDI results with the actual conditions (e.g. fractographic results). To validate and to improve the fatigue crack initiation models and corrosion models.

After teardown operations, the following major information will be obtained and stored in a database:

- Data and results of all inspections, including delivery of signal response data in the form of an electronic database
- Data characterizing the state of damage including:
  - –Fatigue crack distributions, locations, shapes, and sizes
  - –Damage initiation mechanisms and locations
  - –Reconstructed fatigue crack growth histories
- Quantification of corrosion, disbonds, fretting damage at faying surfaces, and other damage
- Descriptions of the crack growth analysis methodologies used
- Results of application of the methodologies as a means to analyze crack growth
- Results of application of the methodologies as a means to predict crack growth
- Description of the methods used to determine the MSD initiation, crack detection, and crack linkup
- Results of the analysis to determine MSD initiation, crack detection, and crack linkup
- Procedure and data from material characterization
- Conclusions and recommendations specific to determination of MSD initiation, crack detection, and crack linkup

According to Bakuckas and Carter, fractography examinations can be accomplished using optical microscope and scanning electron microscope. The extent of fatigue cracking, corrosion, faying surface fretting fatigue, and structural disbanding will be quantified through fractographic examinations. Select fastener holes will be split open to reveal the crack surfaces, and fractographic examinations will be performed to identify, catalog, and document crack initiation sites and mechanisms, crack shapes and sizes, and quality of the fastener hole surface. In addition, the crack growth histories will be empirically reconstructed using striation counts. Crack growth will be backed out to determine equivalent initial flaw size (EIFS) for later analysis validation. Fasteners should be removed without drilling when possible, with care taken to minimize hole damage [Coulter]. For taper-lok fasteners, they must be removed by knocking out with a mallet [Cochran]. A tool is screwed on to the threaded end of the fastener, which prevents the end from being deformed, so that it can pass through the hole without scoring the wall. After the fasteners are removed, the holes are cleaned with solvent using a non-abrasive brush. According to Piascik, Willard and Miller, each rivet can be removed by (1) cut rivet/hole from panel, (2) section rivet/hole specimen using low speed diamond saw. Each rivet half was removed with little or no force, thus exposing the hole inside diameter without disturbing the surface. To open small incipient fatigue cracks located on the rivet hole surface, specimen can be strained opened in a three bend fixture following the slow speed diamond saw cut. This is a type of split open.

Lindsay et al [54] conducted fatigue test based upon a theoretically derived load spectrum for a typical VC10 operating a 3-hour flight plan at an average take-off- weight of 250,000 lb. Fatigue type record was established by British Aerospace. Tear down investigation covered areas include:

The aft face of the front pressure bulkhead

The attachment region of the fuselage keel box to the lower surface of the wing torque box and center section Y-beams

The pressure shell behind the Engineers and Navigators stations and behind the Radio Electronics Equipment Racks

Engine beam assemblies/areas of attachment of engine beam to fuselage

Fuselage lap joints and fuselage circumferential butt joints

The forward freight bay and rear pressure bulkhead

According to Goranson, Boeing has conducted teardown inspections and evaluations of high-time airplanes as a part of a continuing assessment of airplane structure, refer toTable 7.

Table 7:Boeing teardown activities.

Structure	**Year**
707 wing plus center section	1965
707 wing	1968
707 wing plus center section and fuselage	1973
707 empennage	1978
727 forward fuselage	1978
727 wing and empennage	1994
727 fuselage	1996
737 wing plus center section, forward fuselage, and empennage	1987
737 aft fuselage	1988
747 wing and empennage	1989
747 fuselage	1991
777 wing, fuselage, and empennage	1997

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